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Volume IV

ADVANCED METALLIC STRUCTURES: AIR SUPERIORITY FIGHTER WING DESIGN FOR IMPROVED COST, WEIGHT AND INTEGRITY

VOLUME IV BASELINE DAMAGE TOLERANCE EVALUATION

D. F. Davis, et al.

GENERAL DYNAMICS
Convair Aerospace Division
Fort Worth Operation

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WEIGHT AND INTEGRITY

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Tolerance Evaluation

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FOREWORD

The efforts reported herein were sponsored by the Air Force Flight Dynamics Laboratory (AFFDL) under the joint management and technical direction of AFFDL and the Air Force Materials Laboratory, WPAFB, Ohio, 45433. The work was performed under Contract F33615-72-C-2149, Flight Dynamics Laboratory Project Number 486U, "Advanced Metallic Structures: Air Superiority Fighter Wing Design for Improved Cost, Weight and Integrity." Mr. Lawrence R. Phillips of AFFDL is the Air Force Project Engineer.

These studies were performed by the Structural Design Group, Convair Aerospace Division of General Dynamics, Fort Worth Operation with D. F. Davis as the Program Manager. Other principal participants in the program are as follows: R. W. McAnally, Structural Design; E. W. Gomez, Stress Analysis; J. W. Morrow, Fatigue and Fracture Analysis; J. M. Shults, Materials Engineering; T. E. Henderson, Mass Properties; J. D. Jackson, Value Engineering; J. L. McDaniel, Manufacturing Engineering; B. G. W. Yee, Nondestructive Inspection; D. Duncan, Quality Assurance; H. E. Bratton, Information Transfer; and R. L. Jones, Engineering Test Laboratory.

The work was performed from June 1972 to June 1973 and was released for publication June 1973.

This report has been reviewed and is approved.

John C Frishett

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A B S T R A C T

This report describes the preliminary design and analysis for an Advanced Air Superiority Fighter Stores Loaded, Wet Wing Structure. The wing box of the F-111F airplane designed by the Convair Aerospace Division of General Dynamics was used as the baseline vehicle.

A unique design methodology was followed to arrive at three configurations which offer an optimum balance between structural efficiency and technological advancement. This methodology consists of compiling element concepts; integrating them into cross-section drawings; optimizing them in analytical assemblies; and finally preparing full wing box designs. Each step was followed with a detailed evaluation and ranking step which utilized a formal merit rating system. This system permitted the evaluation of numerous concepts and insured that each technical discipline participated in the design selection.

A subsequent program is proposed to evaluate the capability of the selected design to meet the overall program goals of advancing technology without significantly affecting costs. The subsequent program involves additional preliminary design, a development test program, detail design, manufacture, and tests; including static, fatigue, and damage tolerance testing. Information generated during this effort will be disseminated to the Air Force and industry in general through an intensive information transfer effort.

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APPENDIX IX

BASELINE DAMAGE TOLERANCE EVALUATION

IX.1 INTRODUCTION

Utilizing the engineering concept of fracture mechanics as primary technology in providing damage tolerant designs is one of the stated objectives of Contract No. F33615-72-C-2149. Accomplishing this objective is outlined in the original contract technical proposal (FZP-1402) as a fracture control plan. This control plan is directed specifically toward new wing design concepts and involves the following elements:

(1) Material Selection

(2) Damage Tolerance Criteria

(3) Design Support to Insure Damage Tolerance

(4) Fracture Analysis

(5) Risk Assessment Analysis

(6) Manufacturing and Process Control

(7) Quality Assurance

(8) Damage Tolerance Test Program

The original contract does not require that this control plan be applied to the F-111F baseline.

Additional sensitivity and trade studies have been completed to assess the impact of incorporating and applying the new damage tolerance requirements of the proposed version of MIL-STD-1530 (USAF), dated September 1972, and the USAF Damage Tolerance Criteria contained in the latest version of the proposed revision to MIL-A-008866, dated 18 August 1972, to the baseline structure and materials. Elements of a fracture control plan such as described above for the new concepts have been studied for their applicability to the baseline.

The results of these studies provide a quantitative and more thorough understanding of the impact these damage tolerance considerations have on an aircraft structural integrity program.

IX.2 PROGRAM OBJECTIVES AND SCOPE

The basic objective of this study was to provide an updated analysis of the F-111F baseline wing box reflecting the latest proposed Air Force version of damage tolerance criteria. In addition, sensitivity and trade studies were made on the baseline. The effect on allowable stress and service life due to variation in K_{Ic}, da/dN, initial damage assumptions, and service usage were determined. NDI experience, thermal and chemical environment, and the impact of a fracture control plan were studied. Baseline data on inspection experience was compiled. The impact on stress levels and life of varying the residual strength load requirement was determined.

IX.3 PROGRAM DISCUSSION

Utilizing the engineering concept of fracture mechanics as primary technology in providing damage tolerant designs is one of the stated objectives of this program. Accomplishing this objective is outlined in the original contract technical proposal (FZP-1402) as a fracture control plan. This control plan is directed specifically toward new wing design concepts. The original contract does not require that this control plan be applied to the F-111F baseline.

To incorporate and apply the new damage tolerance requirements to the baseline was the subject of an addendum to the basic program. A description of the baseline assessment tasks is given in Technical Proposal FZP-1402, Addendum 1. The purpose of this report is to document the results of this study. The arrangement of paragraphs in this report is identical to that provided in the Air Force statement of work.

The detail damage tolerance requirements addressed by this study are those proposed by the Air Force as a revision to MIL-A-8866. These requirements are summarized in Tables I, II, and III for slow crack growth, fail safe multiple load path, and fail safe crack arrest structure, respectively. In addition, the new fracture control plan elements given in MIL-STD-1530 (USAF) were assessed for impact on the baseline, and baseline inspection experience was compiled.

These studies were performed using the baseline F-111F wing box structure, material, and loads spectra. Preceeding the presentation of study results in subsequent paragraphs, background information not specifically required but necessary to accomplish the baseline damage tolerance assessment is described below.

IX.3.1 Baseline Structure and Stress Distributions

An overall view of the baseline structure is shown in Figure 1. Additional detail of the wing spars is shown pictorially in Figure 2.

Table IV summarizes the four F-111F design conditions used in this study.

Table I

PROPOSED DAMAGE TOLERANCE REQUIREMENTS

SLOW CRACK GROWTH STRUCTURE

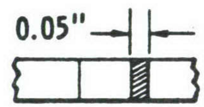
DEGREE OF INSPECTABILITY	FREQUENCY OF INSPECTION	MIN. PERIOD OF UNREPAIRED SERVICE USAGE (F_{XX})	MIN. REQ'D RESIDUAL STRENGTH (P_{XX})	MIN. ASSUMED INITIAL DAMAGE SIZES (a)	MIN. ASSUMED IN-SERVICE DAMAGE SIZES (1)	DAMAGE GROWTH LIMITS
IN FLIGHT EVIDENT	← N/A →					
GROUND EVIDENT	← N/A →					
WALK AROUND VISUAL	SPECIFIED IN CONTRACT DOCUMENTS (10 FLTS. TYPICAL)	5 x FREQ (F_{WV})	P_{WV}	a/Q = 0.10 OR 0.05"  OR SMALLER IF DEMONSTRATED TO .9 P(d) @ 95% C.L.	2" Open thru Crack Unless Detection of Smaller Sizes Demonstrated	1 Shall not grow to critical @ P_{WV} in F_{WV} a Shall not grow to critical @ P_{DM} in F_{DM}
SPECIAL VISUAL	SPECIFIED IN CONTRACT DOCUMENTS (1 YR TYP)	2 x FREQ (F_{SV})	P_{SV}			1 Shall not grow to critical @ P_{SV} in F_{SV} a Shall not grow to critical @ P_{DM} in F_{DM}
DEPOT OR BASE LEVEL	SPECIFIED IN CONTRACT DOCUMENTS (1/4 LIFE-TIME TYP.)	2 x FREQ (F_{DM})	P_{DM}		(a/Q) DM On A/P and Off A/P	1 Shall not grow to critical @ P_{DM} in F_{DM} a Shall not grow to critical @ P_{DM} in F_{DM}
NON INSPECTABLE	N/A	2 LIFETIMES (F_{LT})	P_{LT}		N/A	a Shall not grow to critical @ P_{LT} in F_{LT}

Table II

PROPOSED DAMAGE TOLERANCE REQUIREMENTS

FAIL SAFE - MULTIPLE LOAD PATH STRUCTURE

DEGREE OF INSPECT-ABILITY	FREQUENCY OF INSPECTION	MIN. PERIOD OF UNREPAIRED SERVICE USAGE (F_{XX})	MIN. REQ'D RESIDUAL STRENGTH (P_{XX})	MIN. ASSUMED INITIAL DAMAGE SIZE			MIN ASSUMED IN-SERVICE DAMAGE SIZE	DAMAGE GROWTH LIMITS
				INTACT NEW STRUCTURE (a_1)	REMAINING STRUCTURE: DEPENDENT LOAD PATH (a_2)	REMAINING STRUCTURE: INDEPENDENT LOAD PATH (a_3)		
IN FLIGHT EVIDENT	N/A	RETURN TO BASE (F_{FE})	P_{FE}	$a/Q = .03$ and .02" or Smaller if Demonstrated to .9 P(d) @ 50% C.L.	Failed Load Path Plus $a_1 + \Delta a$ in Adjacent Load Paths or 2" Crack Plus $a_1 + \Delta a$ in Adjacent Load Paths	Failed Load Path Plus $a/Q = .01$ or .01 + Δa in Adjacent Load Paths or 2" Crack Plus $a/Q = .01$ or .01 + Δa in Adjacent Load Paths	a_2 or a_3	a_1 Shall not Grow to Critical @ P_{DM} in F_{DM} a_2 or a_3 Shall not Grow to Critical @ P_{FE} in F_{FE}
GROUND EVIDENT	EVERY FLIGHT	ONE FLIGHT (F_{GE})	P_{GE}					a_1 Shall not Grow to Critical @ P_{DM} in F_{DM} a_2 or a_3 Shall not Grow Critical @ P_{GE} in F_{GE}
WALK AROUND VISUAL	SPECIFIED IN CONTRACT DOCUMENTS (10 FLIGHTS TYPICAL)	5 x FREQ (F_{WV})	P_{WV}					a_1 Shall not Grow to Critical @ P_{DM} in F_{DM} a_2 or a_3 Shall not Grow Critical @ P_{WV} in F_{WV}
SPECIAL VISUAL	SPECIFIED IN CONTRACT DOCUMENTS (ONE YEAR TYPICAL)	2 x FREQ (F_{SV})	P_{SV}					a_1 Shall not Grow to Critical @ P_{DM} in F_{DM} a_2 or a_3 Shall not Grow Critical @ P_{SV} in F_{SV}
DEPOT OR BASE LEVEL	SPECIFIED IN CONTRACT DOCUMENTS (1/4 LIFETIME TYPICAL)	2 x FREQ (F_{DM})	P_{DM}				(a/Q) DM as Specified in 2.3.5	a_1 Shall not Grow to Critical @ P_{DM} in F_{DM} a_2 or a_3 Shall not Grow Critical @ P_{DM} in F_{DM} (a/Q) DM Shall not Grow to Critical @ P_{DM} in F_{DM}
NON INSPECTABLE	N/A	ONE LIFETIME (F_{LT})	P_{LT}				N/A	a_1 Shall not Grow to Critical @ P_{LT} in F_{LT} a_2 or a_3 Shall not Grow to Critical @ P_{LT} in F_{LT}

Table III

PROPOSED DAMAGE TOLERANCE REQUIREMENTS
FAIL SAFE - CRACK ARREST STRUCTURE

DEGREE OF INSPECT-ABILITY	FREQUENCY OF INSPECTION	MIN. PERIOD OF UNREPAIRED SERVICE USAGE (F_{XX})	MIN. REQUIRED RESIDUAL STRENGTH (P_{XX})	MIN ASSUMED INITIAL DAMAGE SIZE		MIN. ASSUMED IN-SERVICE DAMAGE SIZE (1)	DAMAGE GROWTH LIMITS
				INTACT NEW STRUCTURE a_1	IN REMAINING STRUCTURE a_2		
IN FLIGHT EVIDENT	N/A	RETURN TO BASE (F_{FE})	P_{FE}	a/Q • 0.03 0.02" or Smaller if Demonstrated to .9 P(d) @ 50% C.L.	2 Cracked Skin Panels Plus Failed Central Stringer (or equivalent)	2 Cracked Skin Panels Plus Failed Central Stringer (or Equivalent)	a_1 Shall not Cause Initial Rapid Propagation @ P_{DM} in F_{DM} 1 Shall not Cause Complete Failure @ P_{FE} in F_{FE}
GROUND EVIDENT	EVERY FLIGHT	ONE FLIGHT (F_{GE})	P_{GE}				a_1 Shall not Cause Initial Rapid Propagation at P_{DM} in F_{DM} 1 Shall not Cause Complete Failure @ P_{GE} in F_{GE}
WALK AROUND VISUAL	SPECIFIED IN CONTRACT DOCUMENTS (10 FLIGHTS TYPICAL)	5 x FREQ (F_{WV})	P_{WV}			a_2 or 2" or Greater through Crack in Skin at Failed Stringer whichever is Applicable Smaller Crack if Demonstrated	a_1 Shall not Cause Initial Rapid Propagation @ P_{DM} in F_{DM} 1 Shall not Cause Complete Failure @ P_{WV} in F_{WV}
SPECIAL VISUAL	SPECIFIED IN CONTRACT DOCUMENTS (ONE YEAR TYPICAL)	2 x FREQ (F_{SV})	P_{SV}				a_1 Shall not Cause Initial Rapid Propagation @ P_{DM} in F_{DM} 1 Shall not Cause Complete Failure @ P_{SV} in F_{SV}
DEPOT OR BASE LEVEL	SPECIFIED IN CONTRACT DOCUMENTS (1/4 LIFETIME TYPICAL)	2 x FREQ (F_{DM})	P_{DM}			(a/Q) DM as Specified in 2.3.5 or a_2	a_1 Shall not Cause Initial Rapid Propagation @ P_{DM} in F_{DM} 1 Shall not Cause Complete Failure @ P_{DM} in F_{DM}

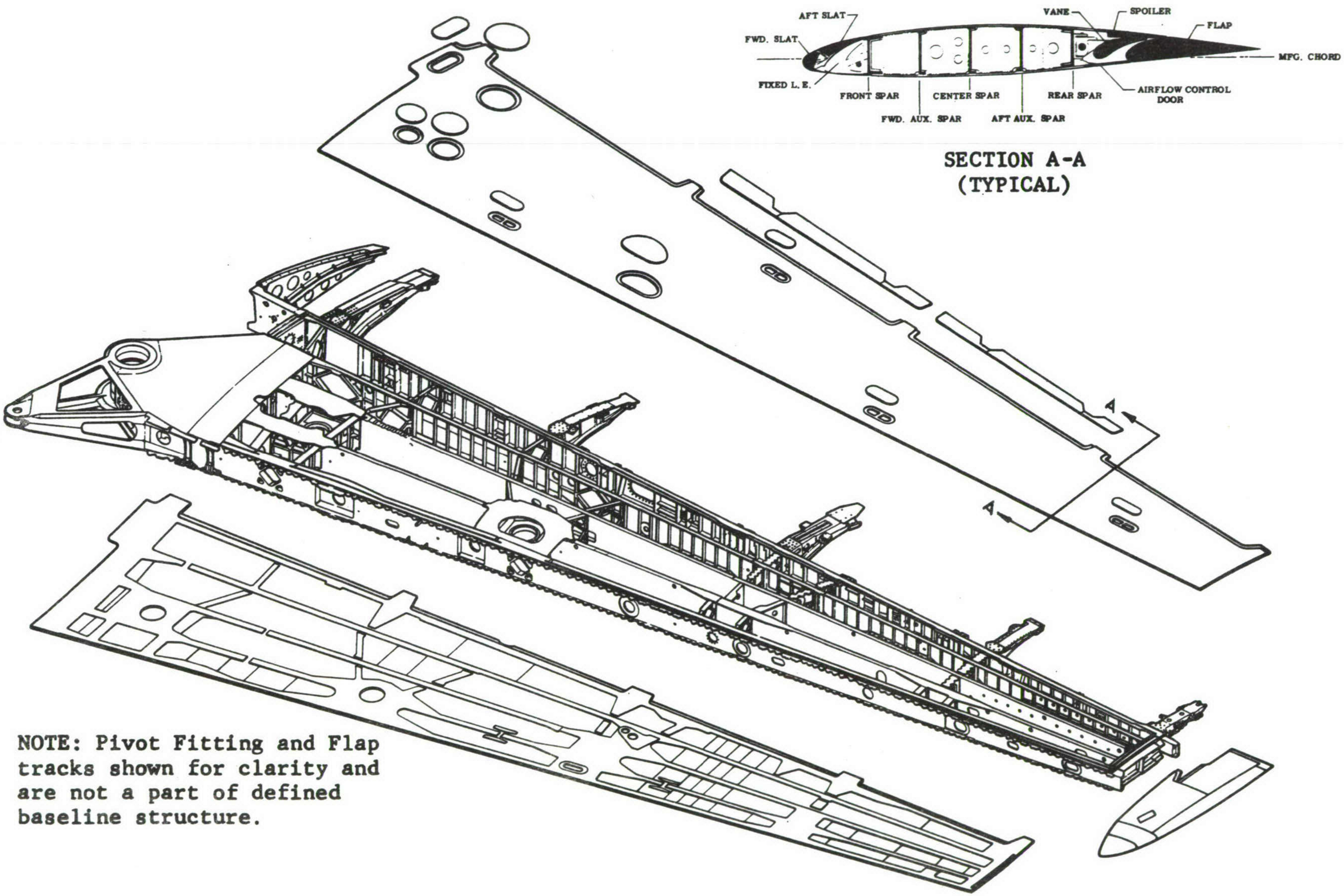


Figure 1 F-111 Baseline Wing Box Structure

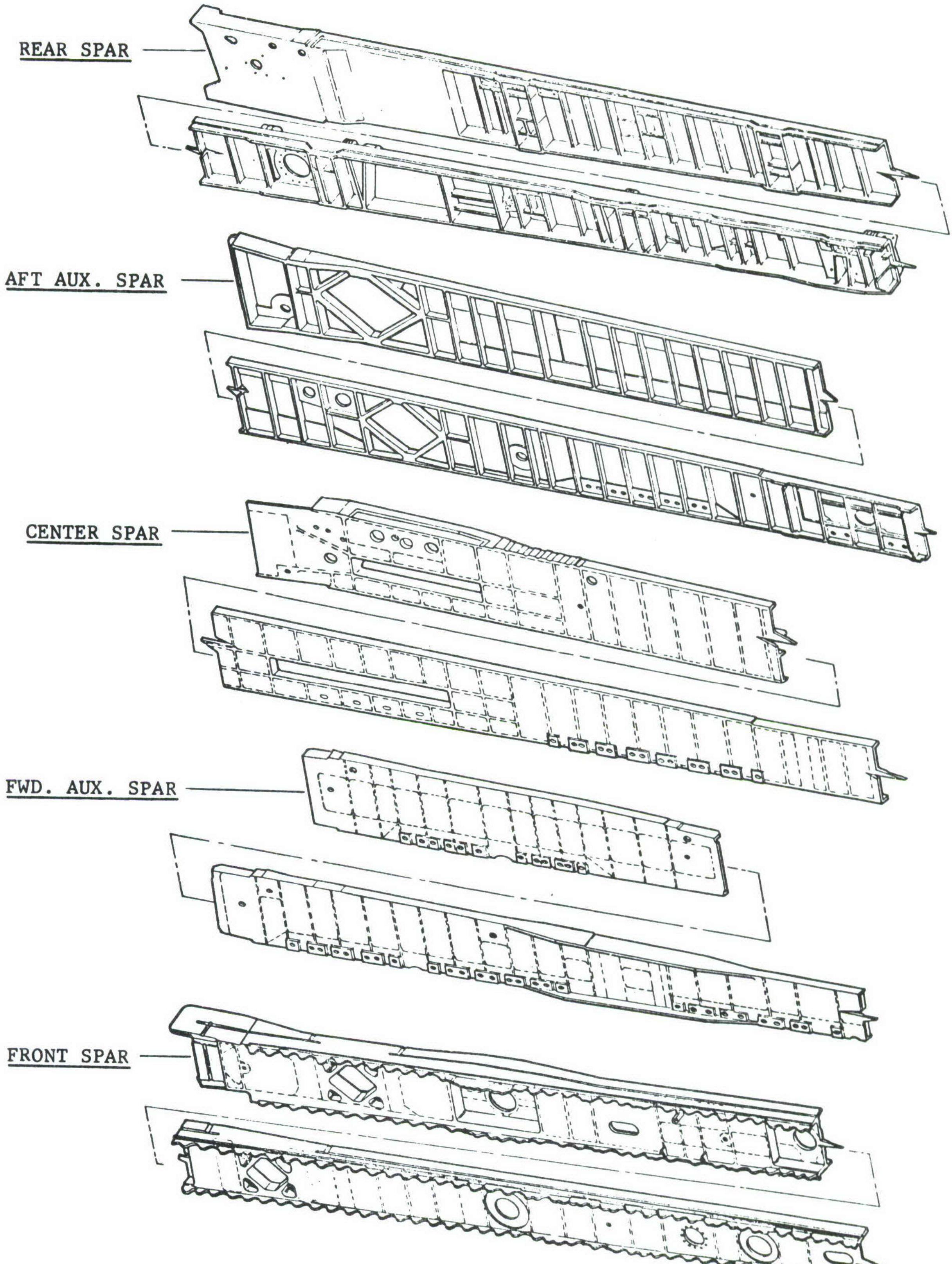


Figure 2 Typical Detail of Baseline Wing Spars

Table IV

SUMMARY OF DESIGN CONDITIONS

① F101A (M = 300 KCAS, $\Lambda = 16^o$, S.L., $n_Z = 4.124$)

- o Ultimate $BM_p = 26.49 \times 10^6$ in. lbs.
- o Limit $BM_p = 17.66 \times 10^6$ in. lbs.

② F400A ($M = 1.05$, $\Lambda = 45^o$, $h = 2000$ ft, $n_Z = 7.33$)

- o Ultimate $BM_p = 29.28 \times 10^6$ in. lb.
- o Limit $BM_p = 19.52 \times 10^6$ in. lb.

③ F401A ($M = 1.05$, $\Lambda = 45^o$, $h = 8000$ ft, $n_Z = - 3.00$)

- o Ultimate $BM_p = - 14.87 \times 10^6$ in. lb.
- o Limit $BM_p = - 9.91 \times 10^6$ in. lb.

④ F702A ($M = 1.40$, $\Lambda = 72.5^o$, $h = 17{,}500$ ft, $n_Z = - 3.00$)

- o Ultimate $BM_p = - 12.45 \times 10^6$ in. lb.
- o Limit $BM_p = - 8.30 \times 10^6$ in. lb.

For each of the spanwise center spar stations (C.S.S.) shown in Figure 3 plots indicating baseline lower wing skin stress distributions are given in Figure 4. Similar stress data for the lower spar caps is given in Figures 5 and 6.

The F-111F wing box is classified as slow crack growth structure. The lower wing skin is the only part whose failure would result in catastrophic loss of the aircraft. The five spar design configuration provides overall multiple load path capability. However, consideration of flaws in the lower spar caps is considered important because of their direct mechanical attachment to the lower skin. The presence of an assumed flaw in the spar caps is not easily detectable without removal of the upper skin, and such a flaw could initiate damage to the lower skin at the bolt hole attachments. The criteria, in fact, specifies that mating parts which experience the same manufacturing operations (e.g., bolt holes) will each be assumed to have similar damage. Therefore, in addition to the bolt hole flaw, a part through flaw was assumed in a typical spar cap thickness of 0.25 inches. See Figure 7.

The lower skin thickness is tapered, and the spanwise stress distributions are fairly uniform outboard of the splice to about C.S.S. 237 as shown by the stress plots in Figures 4 and 6. Lower surface stresses are highest in the area of C.S.S. 140 to C.S.S. 237. A part through flaw was assumed in the 0.611 in. thick lower skin at C.S.S. 140 to typically represent this area. Taper-lok fasteners 5/16 in. in diameter are used for skin-to-spar cap connections in this area, so bolt hole flaws were assumed to exist in 5/16 in. diameter holes. See Figure 7. The maximum thickness of the lower skin in the area just outboard of the lower skin-to-pivot fitting lower plate is approximately 1.30 inches. A part through surface flaw was also assumed for this location to indicate the effect of thickness.

The wing-to-pivot fitting splice is accomplished with four rows of 5/8 in. diameter taperloks. However, the splice is not defined as part of the baseline and was not included in these studies. In addition, stresses could be decreased in the area of the splice by adding weight (thickness) over a localized area with very little total weight penalty to the entire wing box.

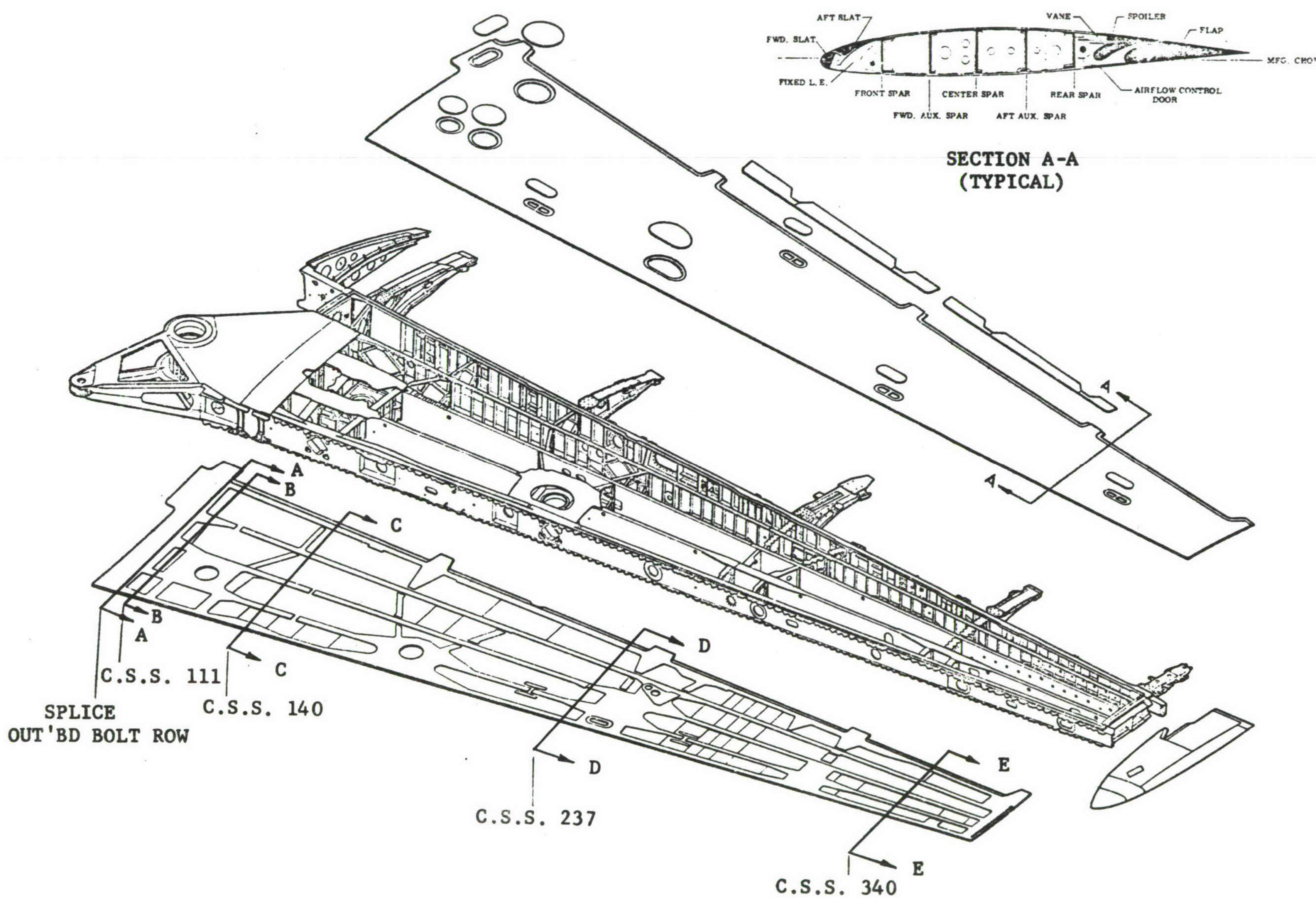


Figure 3 Center Spar Station Locations for Lower Wing Skin Stress Distributions

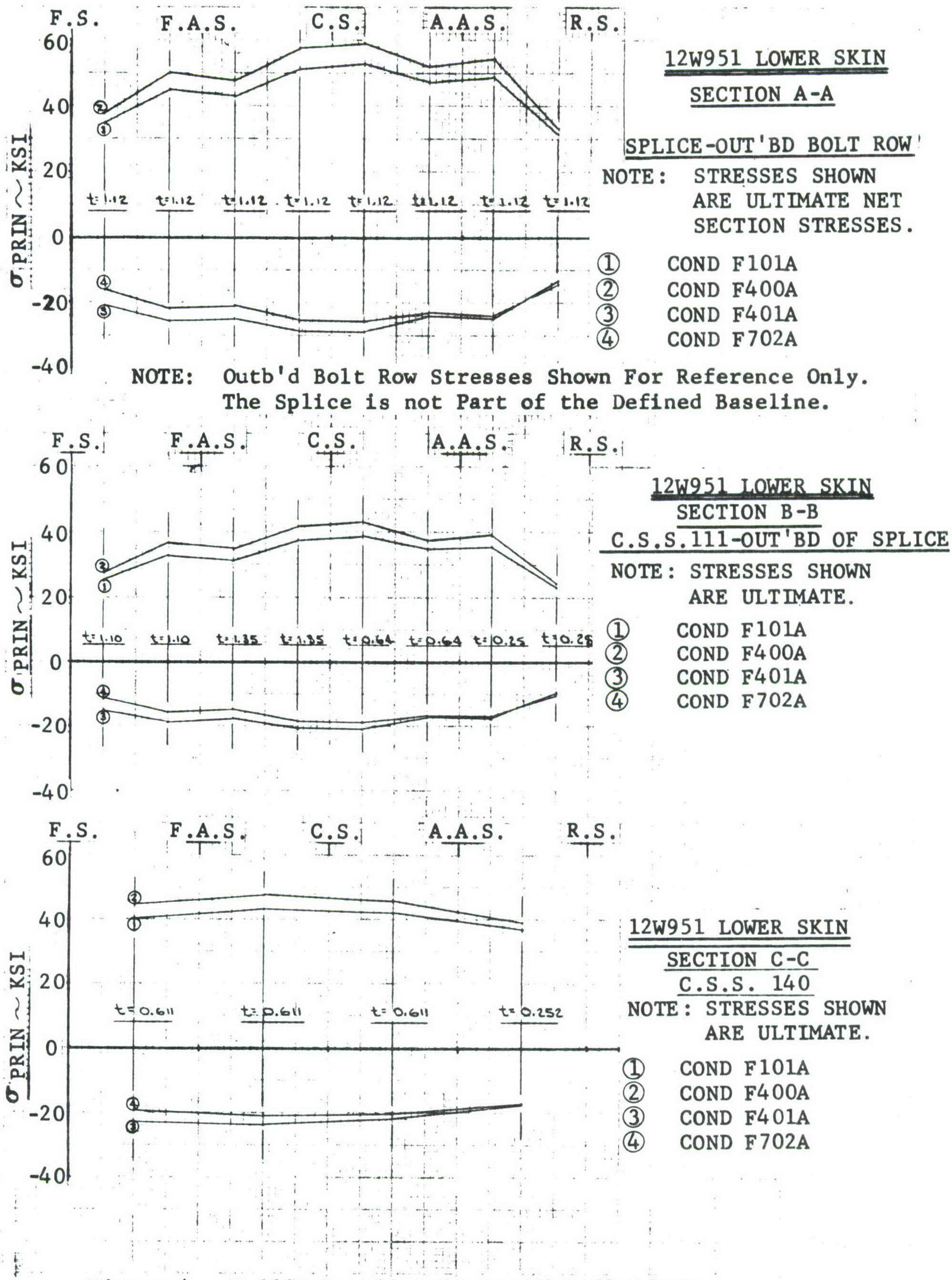
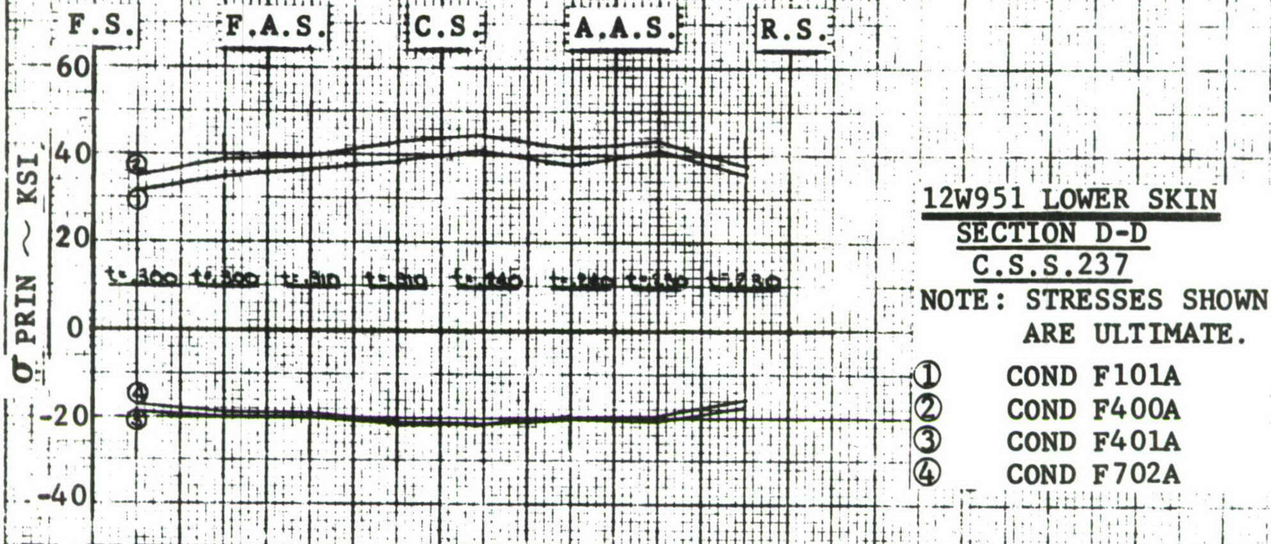


Figure 4 F-111F Baseline Stress Distributions



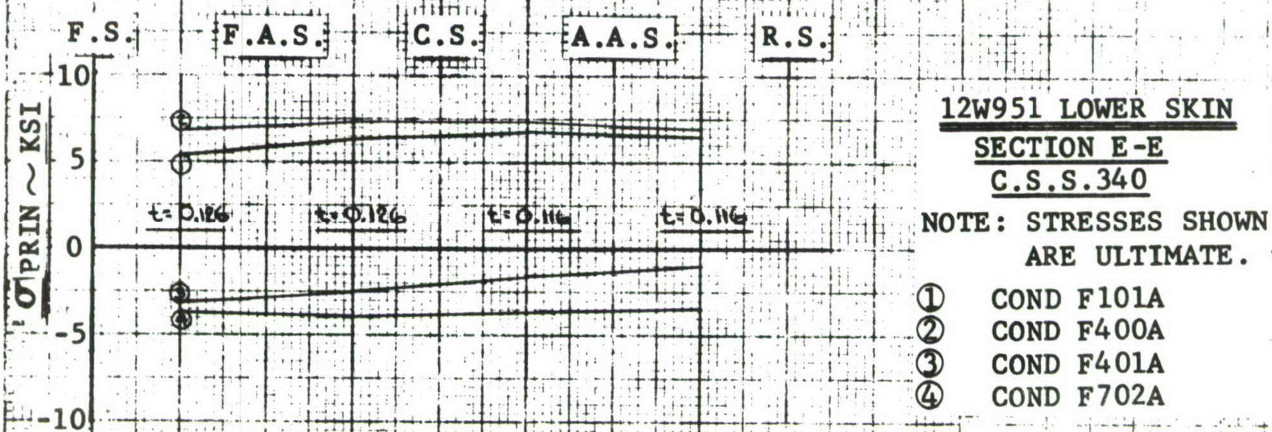


Figure 4 (Cont'd) F-111F Baseline Stress Distribution

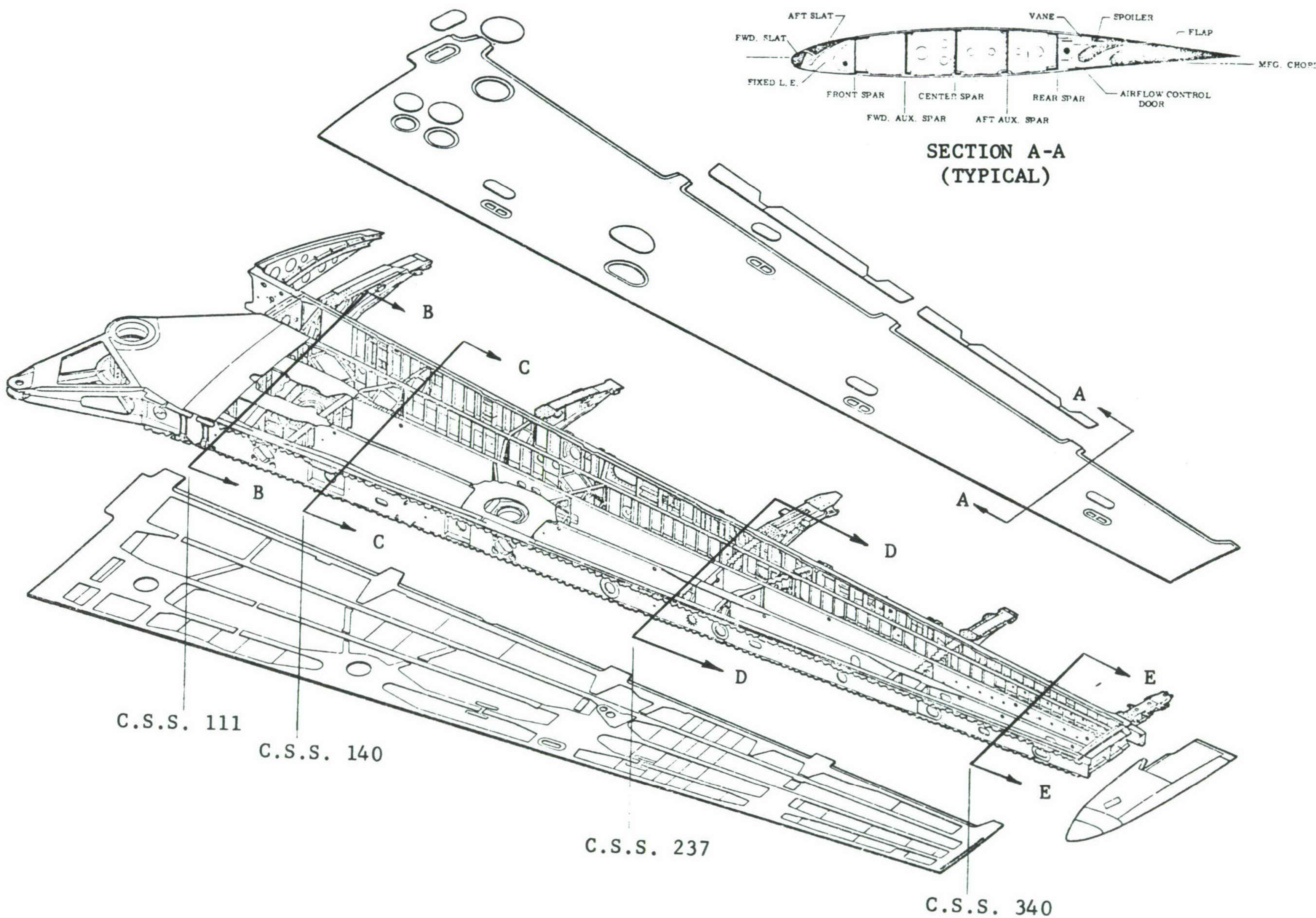


Figure 5 Center Spar Station Locations For Lower Spar Cap Stress Distributions

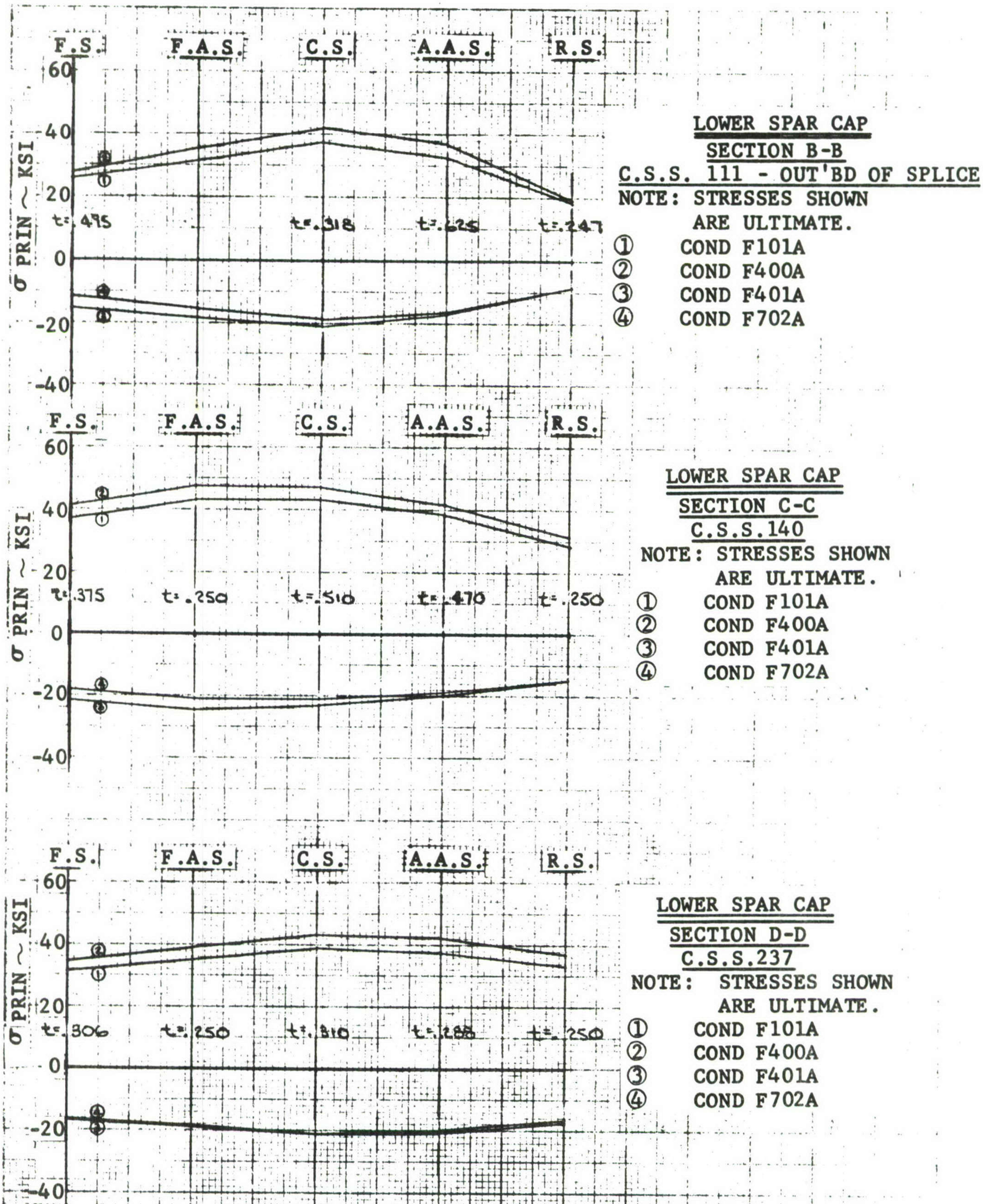


Figure 6 F-111F Baseline Stress Distributions

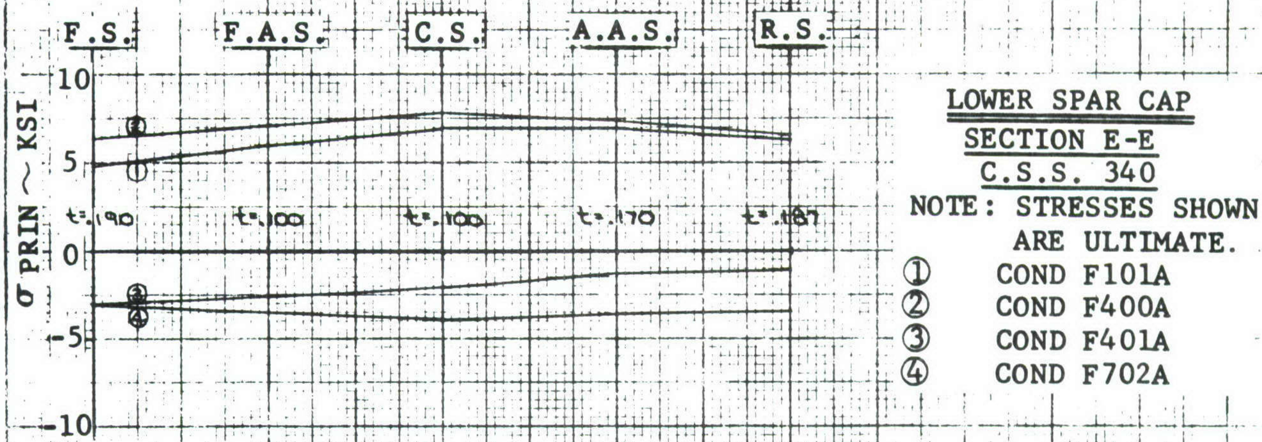
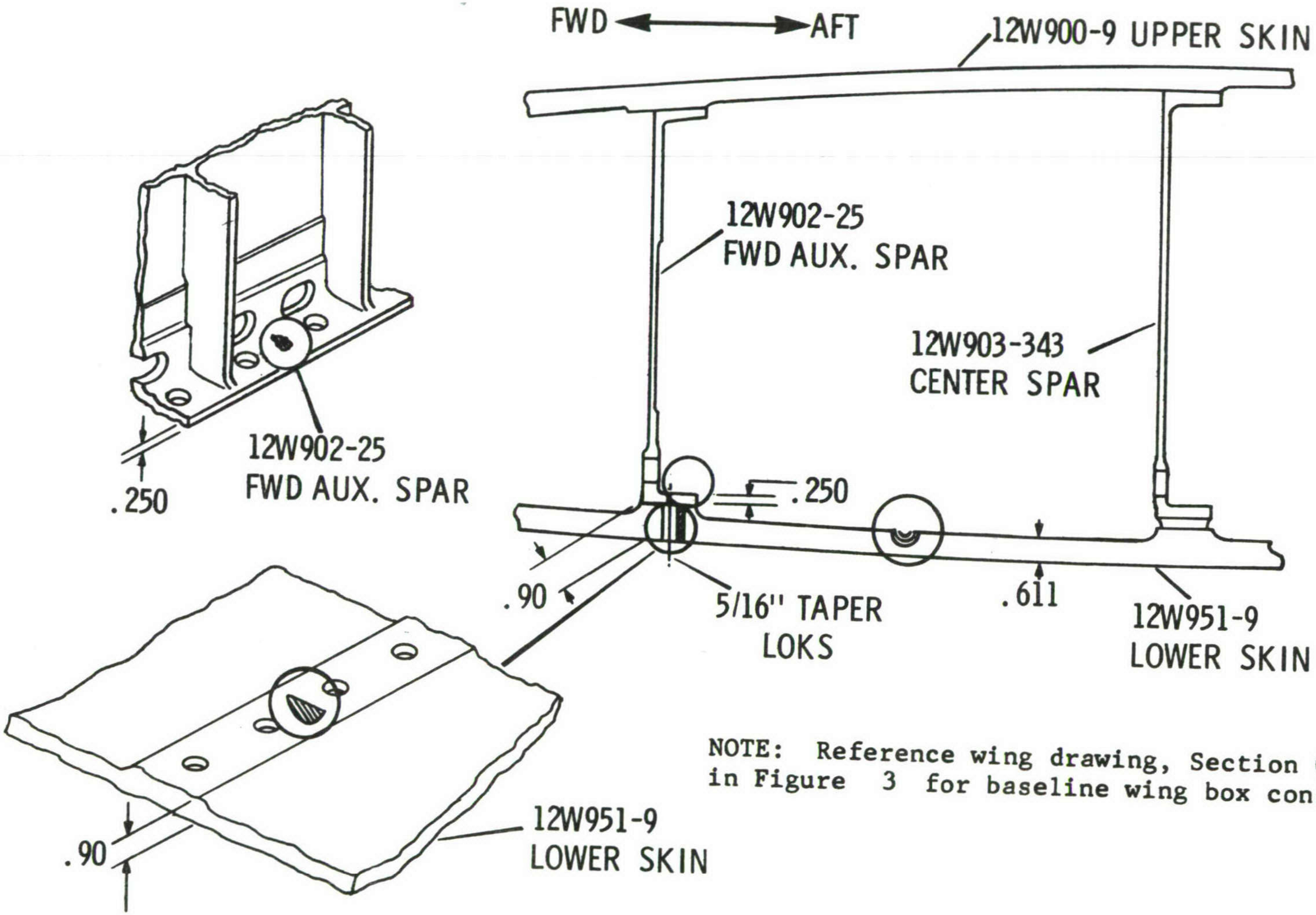


Figure 6 (Cont'd) F-111F Baseline Stress Distributions



NOTE: Reference wing drawing, Section C-C, in Figure 3 for baseline wing box configuratio

Figure 7 F-111F Baseline Flaw Locations at C.S.S. 140 Cross Section

The upper surface (wing skin and spar cap) of the baseline wing is not considered a design problem for two reasons, (1) tensile stresses are low in magnitude because the upper surface is designed primarily for compression buckling which resulted in substantial load carrying area, and (2) tensile stresses in the upper surface occur primarily from negative maneuvers which comprise very little of the time prorated to F-111 maneuvers. Consequently, there are very few occurrences of tensile loading in the upper surface.

IX.3.1.1 Crack Growth Analysis Assumptions

The following assumptions and groundrules were used for crack growth analysis:

- Stress intensity models as shown in Figure 8.

  The bolt hole models account for geometric stress concentration at the edge of the hole. The maximum value of stress concentration has been defined as the ratio σ_{ys}/σ_{max}, where σ_{ys} = 58 ksi for 2024-T851 and σ_{max} is the maximum stress in the fatigue spectrum. This definition is based on the reasoning that peak stresses are limited by plastic flow, e.g., G_{max} (σ_{max}) = σ_{ys}. The bolt hole models do not account for the effect of installed fastener systems.

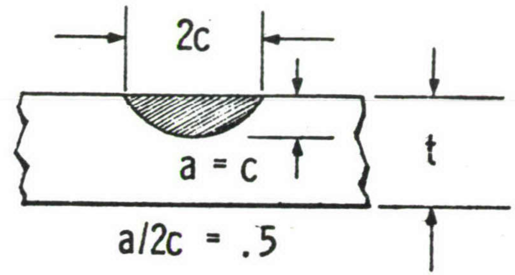
  The semi-circular shape has been assumed for analysis involving part through surface flaws based on a/2c measurements taken during the 2024-T851 spectrum/environmental tests described under a separate heading below. These measurements indicated a flaw shape of .5 and greater for most of the test history on each specimen.

  Experience with flaw shapes measured on post-failure fracture surfaces of D6ac steel specimens tested during the F-111 Recovery Program, indicated that flaws having an initial a/2c = .1 would grow rapidly in the depth direction, tending to form a semi-circular shape early in the spectrum loading. With this in mind, additional analytical studies were performed in which an assumed initial flaw, having an initial shape of

- o Surface Flaw - Part Through

$$K = M_K \, 1.1\sigma \sqrt{\pi a/Q} \text{ , or}$$

$$K = M_K \frac{1.1\sigma\sqrt{\pi a}}{\sqrt{\phi^2 - 0.212(\sigma/\sigma_{ys})^2}}$$



- o Surface Flaw - Through the Thickness

$$K = \sigma\sqrt{W \text{ Tan } \left(\frac{\pi a}{W} + \frac{K^2}{2W\sigma_{ys}{}^2}\right)}$$

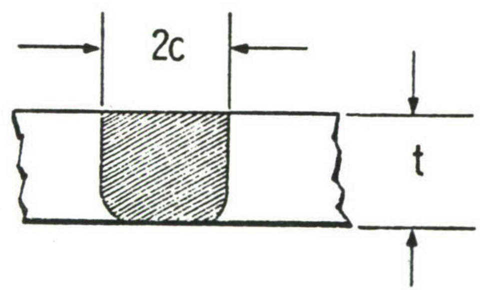


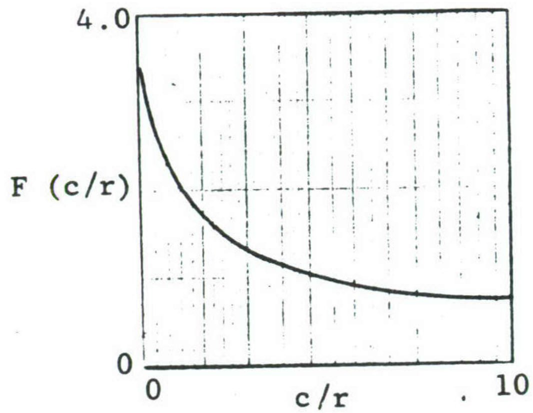
Figure 8 Stress Intensity Models

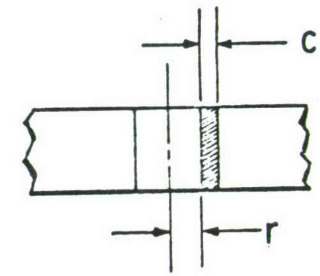
o **Bolt Holes - Through the Thickness**

$$K = \sigma\sqrt{\pi c} \quad F(c/r)$$

Where $F(c/r)_{max} = \sigma_{ys}/\sigma_{max}$

by definition





BOWIE MODEL

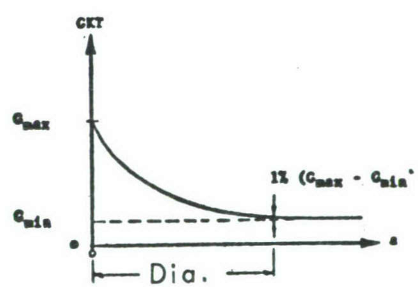
o **Bolt Holes - Semicircular Corner Crack**

$$K = 1.2\sigma\sqrt{\pi a/Q}\left(GKT\right)$$

$$Q = \pi/2$$

$$GKT = G_{min} + (G_{max} - G_{min})\ \mathrm{Exp}\left[\mathrm{Ln}(.01)\frac{a}{DIA}\right]$$

$G_{min} = 1.0$, $G_{max} = \sigma_{ys}/\sigma_{max}$ by definition



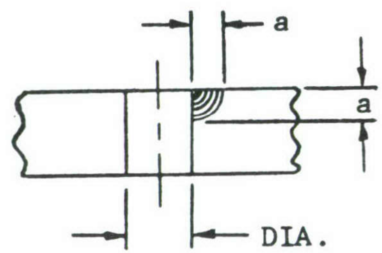


Figure 8 (cont'd) Stress Intensity Models

a/2c = .1, was allowed to grow in depth (a) while holding the surface length (2c) constant. The shape was therefore varied as the flaw propagated, i.e., the backface correction (M_k) and the elliptic integral (ϕ) varied as a/2c changed. This variation in shape was continued until the flaw depth was equal to one-half the flaw length. The shape for growth beyond this point was then assumed to be semi-circular. The results of this additional study are presented separately in Section IX.5. There was no significant difference in the allowable stresses established using a/2c = .1 or a/2c = .5 as the initial shape, particularly for the 4000 and 8000 hour periods of unrepaired service usage.

- Backface corrections (M_K) for part through flaws were based on curves given in AFFDL-TR-70-107.

  When required, transition from part through surface flaw calculations to through the thickness surface flaw calculations is automatic within the computer program used for crack growth analysis. Convair Aerospace flaw growth computer program TD9 was used for all crack growth analyses in these studies.

- m = 1.6 is used in the Wheeler retardation model and is based on analytical correction of the test results described under that paragraph heading below.

- Crack growth (da/dn) data used for analysis is based on data given in AFML-TR-66-291 for 2024-T851 plate, R = 1/3, 310 cpm, RT air. The data is shown in Figure 9. Upper and lower bounds assumed for this data are also indicated on the figure. The Forman equation was used to account for other R values.

  The baseline chemical environment has been established as water saturated JP-4 fuel. However, the da/dn data given in AFML-TR-66-291 was compared to JP-4 data for 2024-T851 available in MDC A0874 (F-15) and found to be conservative. Comparison of the AFML-TR-66-291 data with data generated at Convair Aerospace for 2024-T852 forgings in sump water (B-1 program) also indicated that the AFML data was conservative.

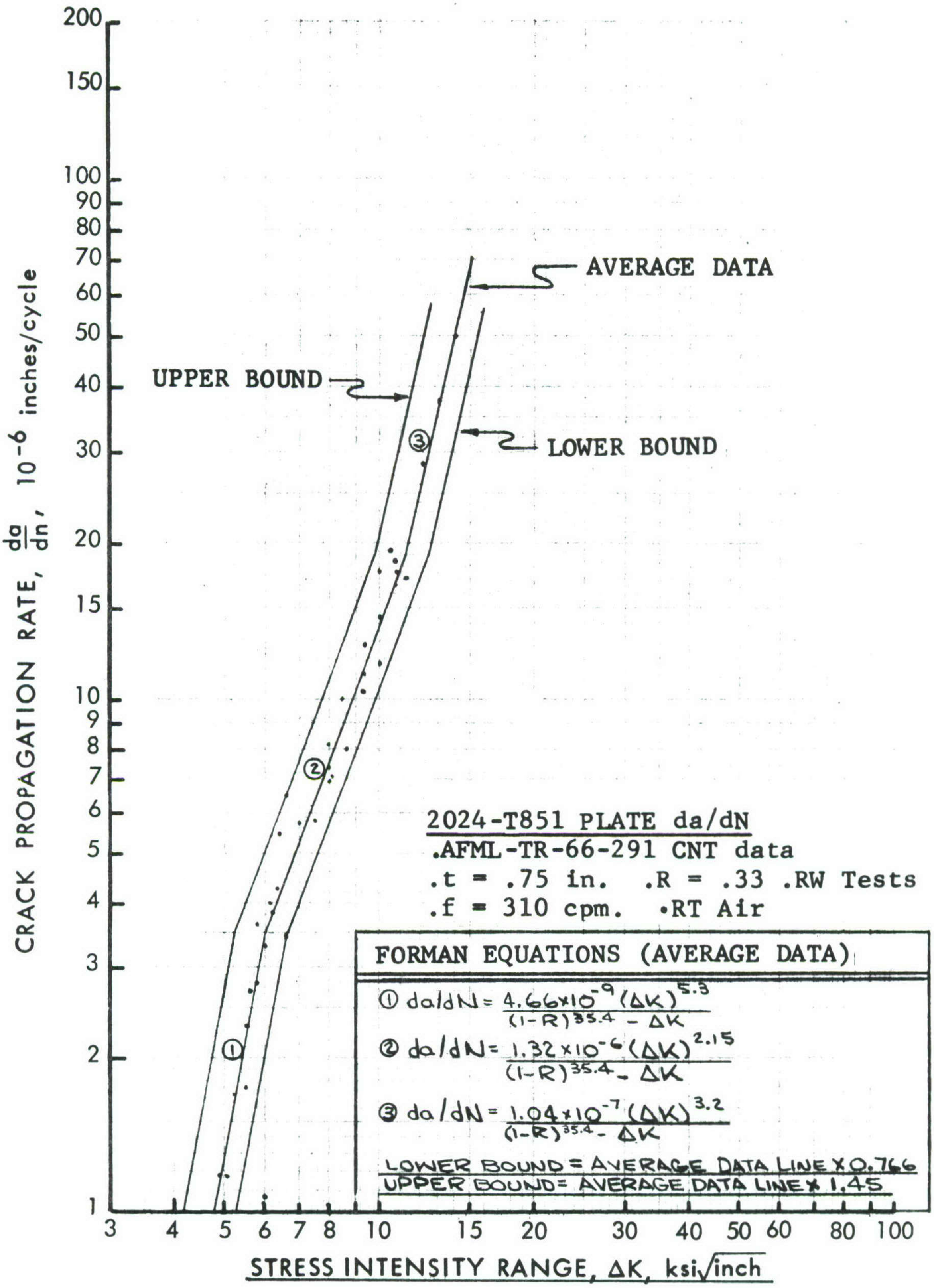


Figure 9 - Crack Growth Data

- Fracture toughness (K_{Ic}) data was selected according to the studies shown in Table V . The lower, mid-point, and upper bounds to the data given in the table for 2024-T851 plate, L-T direction, were used in this study.

- Limit load was used in all cases to calculate critical flaw sizes except in connection with the studies of residual strength determined from load exceedance data (P_{xx}) described separately in paragraph IX.3.10. This decision to use limit load was made so that the basic analytical work could be started prior to complete development of load exceedance residual strength loads. The final determination of these loads for the baseline is now complete and indicates that the use of limit load is conservative. See paragraph IX.3.10.

- The repeated loads spectra are representative of baseline usage (both "mild" and "severe") and reflect the exceedance data in MIL-A-8866A, dated March 1971. The load spectra is randomized and applied in 200 hour block increments. The "mild" usage spectra is reflected only in the work presented in paragraph IX.3.3. (See paragraph IX.3.3 for a definition of "mild" and "severe" usage.)

- The wing bending moment spectrum at the pivot is used as the basis for analysis.

  - Max. Spectrum BM @ pivot = 15.55×10^6 in.lbs.

  - Limit BM @ pivot = 19.52×10^6 in. lbs. (Reference design condition F400A)

  - Ultimate BM @ pivot = 29.3×10^6 in. lbs. (Reference design condition F400A).

IX.3.1.2 Spectrum/Environmental Test Results

Spectrum/environmental tests applicable to the baseline were conducted to provide a basis for establishing a value for the retardation exponent, m, used in the Wheeler Retardation model.

Table V

FRACTURE TOUGHNESS OF ALUMINUM ALLOYS USED IN WING BOX OF F-111F

(KSI $\sqrt{in.}$)

Ref.: MCIC-HB-01 for Code to Specimen Orientation and Crack Propagation Directions

Wing Skins and Intermediate Spars:

	L-T	T-L	S-L
Upper Bound	26	24	21
Average	23	21	17
Lower Bound	20	18	15

2024-T851 Plate 1.0 Through 3.0 inches thick.

Front and Rear Spars:

Short transverse tensile tested 2024-T851 Plate >3.01 inches thick (also known as 2124-T851).

	L-T	T-L	S-L
Upper Bound	33	28	26
Average	26	24	22
Lower Bound	21	19	17

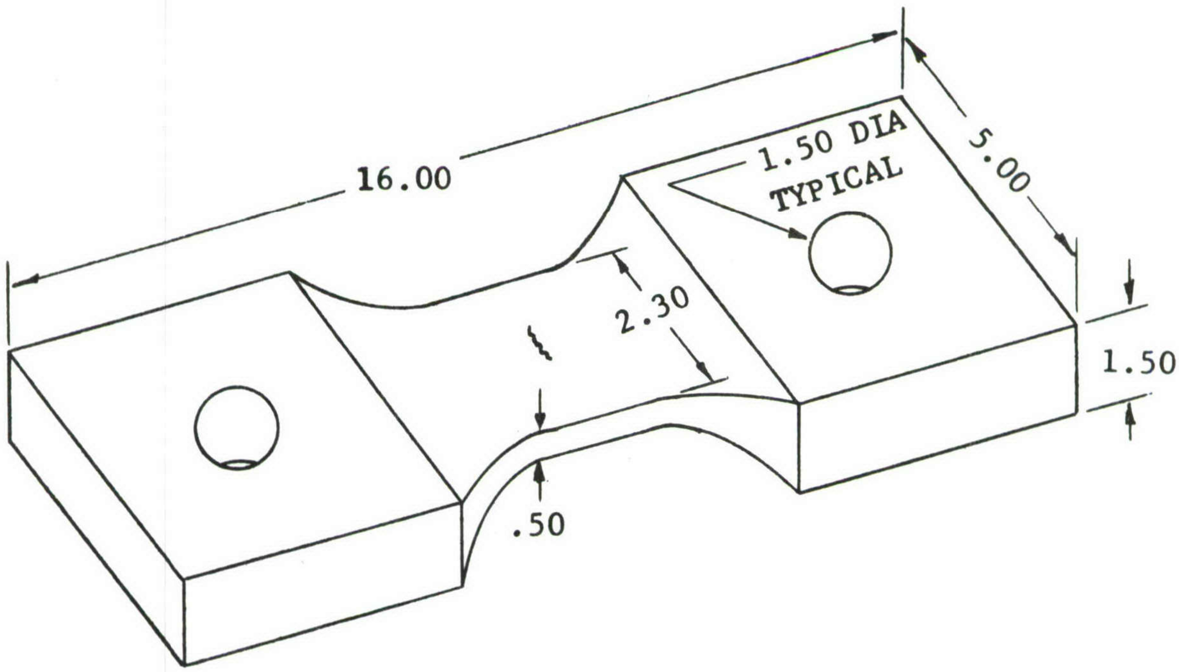
These values are based on review of all data available on 1/31/73.

MIL-HDBK-5B
Alcoa Green Letter on 2124 GL 217 (9-70)
CAD/FW IRAD Studies
Alcoa Contract (F33615-71-C-1571)
MCIC-HB-01

Four 2024-T851 aluminum surface flaw specimens were tested concurrently using a test spectrum reflecting baseline severe usage (Phase I and II Training):

Specimen 24-5 Dry Air
Specimen 24-6 JP-4
Specimen 24-7 JP-4
Specimen 24-8 JP-4

Specimen dimensions are shown in the sketch. The test spectrum is given in Table VI.



The three JP-4 specimens failed under spectrum loading. The dry air specimen was statically pulled. Post failure measurements of crack depth, a, were made from Faxfilm replicas of each fracture surface. Continuous measurement of crack depth was not possible over the entire spectrum loading history due to the presence of intermittent marker bands and the complexity of the growth patterns. However, sufficient measurements of repetitive patterns within groups of blocks were made so that the data could be plotted on log-log paper as (Δa/block) versus crack depth, (a), for three of the four specimens. Specimen 24-7 developed a separate second flaw early in the test program. The data from this specimen was not considered valid and is not included.

Table VI

ADP WING
2024-T851 RANDOM TEST SPECTRUM

	SMIN	SMAX	CYCLES	RATE
1	4.44	11.89	18.0	6.0
2	5.10	14.82	543.0	6.0
3	4.47	10.43	505.0	6.0
4	3.55	14.01	13.0	6.0
5	0.0	5.15	1.0	6.0
6	5.91	14.56	2.0	6.0
7	2.54	12.88	138.0	6.0
8	2.32	10.29	35.0	60.0
9	0.0	5.15	1.0	6.0
10	0.0	3.05	1.0	6.0
11	4.47	11.15	416.0	6.0
12	2.54	10.00	193.0	6.0
13	4.50	11.47	989.0	6.0
14	4.42	11.36	51.0	6.0
15	2.46	9.07	301.0	60.0
16	0.03	14.16	22.0	60.0
17	0.0	11.49	1.0	180.0
18	0.31	6.82	1374.0	60.0
19	0.05	5.10	13.0	6.0
20	5.91	14.14	4.0	6.0
21	2.33	7.58	6.0	60.0
22	5.91	11.87	18.0	6.0
23	0.0	7.26	21.0	60.0
24	5.07	17.22	31.0	6.0
25	5.10	15.34	195.0	6.0
26	5.15	12.25	38.0	6.0
27	5.10	16.70	15.0	6.0
28	4.50	17.59	97.0	6.0
29	6.76	16.73	1.0	6.0
30	3.53	13.51	2.0	6.0
31	6.77	16.23	2.0	6.0
32	5.07	22.37	5.0	6.0
33	5.15	11.24	180.0	6.0
34	2.32	8.76	178.0	60.0
35	4.42	10.64	20.0	6.0
36	2.54	15.63	53.0	6.0
37	7.76	13.95	11.0	6.0
38	1.04	8.91	4.0	60.0
39	7.66	18.71	2.0	6.0
40	4.47	9.49	381.0	6.0
41	6.42	15.16	2.0	6.0
42	6.30	14.24	1.0	6.0
43	3.55	17.25	3.0	6.0
44	6.77	15.33	4.0	6.0
45	1.04	7.26	10.0	60.0

NOTE: SMIN and SMAX are in KSI.CYCLES given per 200-HOUR BLOCK.RATE is CPM.

Table VI (Cont'd)

ADP WING
2024-T851 RANDOM TEST SPECTRUM

	SMIN	SMAX	CYCLES	RATE
46	6.77	11.60	36.0	6.0
47	5.10	10.12	686.0	6.0
48	4.50	16.33	533.0	6.0
49	6.30	12.72	4.0	6.0
50	5.15	10.95	353.0	6.0
51	2.54	20.72	5.0	6.0
52	0.50	9.06	363.0	60.0
53	0.0	20.46	6.0	6.0
54	3.55	9.25	32.0	6.0
55	0.05	12.68	19.0	60.0
56	0.49	6.64	1376.0	60.0
57	0.50	13.30	102.0	60.0
58	0.31	8.76	371.0	60.0
59	5.12	15.68	4.0	6.0
60	5.07	24.25	2.0	6.0
61	0.31	10.29	114.0	60.0
62	5.07	13.46	75.0	6.0
63	0.0	4.42	1.0	6.0
64	0.0	5.64	67.0	60.0
65	2.54	7.50	239.0	6.0
66	4.42	10.09	37.0	6.0
67	0.32	11.05	102.0	60.0
68	4.50	18.92	7.0	6.0
69	2.45	13.30	20.0	60.0
70	0.0	8.91	18.0	60.0
71	0.0	22.91	2.0	6.0
72	2.53	24.14	1.0	6.0
73	0.0	13.90	1.0	180.0
74	3.52	9.48	3.0	6.0
75	0.05	10.64	63.0	60.0
76	5.07	10.87	130.0	6.0
77	3.55	16.48	6.0	6.0
78	5.10	15.76	104.0	6.0
79	0.0	4.42	2.0	6.0
80	5.07	20.27	13.0	6.0
81	0.0	12.70	1.0	180.0
82	7.76	18.48	3.0	6.0
83	0.03	7.58	300.0	60.0
84	2.32	11.03	29.0	60.0
85	4.45	12.52	148.0	6.0
86	4.52	18.16	26.0	6.0
*87	2.51	25.19	0.05	6.0
88	5.93	13.38	6.0	6.0
89	3.55	11.62	26.0	6.0
90	2.58	18.35	19.0	6.0

NOTE: SMIN and SMAX are in KSI.CYCLES given per 200-HOUR BLOCK.RATE is CPM.

* Apply LL 87 once every 20 Blocks.

Table VI (Cont'd)

ADP WING
2024-T851 RANDOM TEST SPECTRUM

	SMIN	SMAX	CYCLES	RATE
91	7.66	13.75	7.0	6.0
92	6.76	13.69	11.0	6.0
93	2.46	11.18	64.0	60.0
94	6.29	15.52	1.0	6.0
95	2.33	14.16	1.0	60.0
96	5.10	12.94	682.0	6.0
97	5.12	16.52	1.0	6.0
98	0.03	5.15	5.0	6.0
99	0.C	5.10	2.0	6.0
100	4.52	21.21	1.0	6.0
101	0.C	17.71	52.0	6.0
102	6.95	12.36	10.0	6.0
103	4.47	11.66	279.0	6.0
104	6.97	16.95	2.0	6.0
105	0.5C	11.19	111.0	60.0
106	4.5C	17.27	280.0	6.0
107	4.50	14.39	810.0	6.0
108	5.15	12.12	74.0	6.0
109	0.C	5.15	2.0	6.0
110	1.10	14.71	1.0	180.0
111	0.C	5.10	4.0	6.0

NOTE: SMIN and SMAX are in KSI.CYCLES given per 200-HOUR BLOCK.RATE is CPM.

A least squares line was used to fit the (Δa/block) vs. (a) data. The resulting equations for these lines were then used to plot crack growth curves for specimens 24-5, 24-6, and 24-8. See Figures 10 through 12.

Analytical correlation of the crack growth test data is shown on the crack growth curves of Figures 13 through 15. The lower bound value of "m" (based on specimen 24-8) was 1.6 when correlation was started at the smaller flaw depths of approximately 0.08". The analytical growth for m = 1.6 is conservative at the upper end of the growth curve.

Correlation of the upper part of the test data curves (flaw depths equivalent to a/Q = .1) indicates the "m" value is in excess of 2.0. However, the more conservative lower bound value of 1.6 was chosen for analysis. Measurements of a/2c are also indicated on the growth plots. Based on these measurements an a/2c = .5 was used for analysis in these studies.

Fracture toughness specimens were tested from the same material used for fabrication of the surface flaw specimens. Resulting K_{Ic} values were:

24.9 LT direction
26.1 LS direction

IX.3.1.3 Baseline Fracture Design Allowable Curves

The analytical tasks in these studies required development of enough data to assess the impact of the various parameters involved in flaw growth analysis on design allowable stresses and life. These tasks were accomplished by generating a substantial quantity of flaw growth analyses which enabled the development of fracture design allowable curves. These curves indicate the interaction between design allowable stress, initial flaw size, and flaw growth residual life. The curves were developed so that the impact of variations in the analysis variables (K_{Ic}, da/dn, flaw type, usage, etc.) can be quickly assessed. Design allowable curves were subsequently developed according to the analysis matrix shown in Table VII. The resulting curves were used directly or cross-plotted to provide the required sensitivity and trade study results.

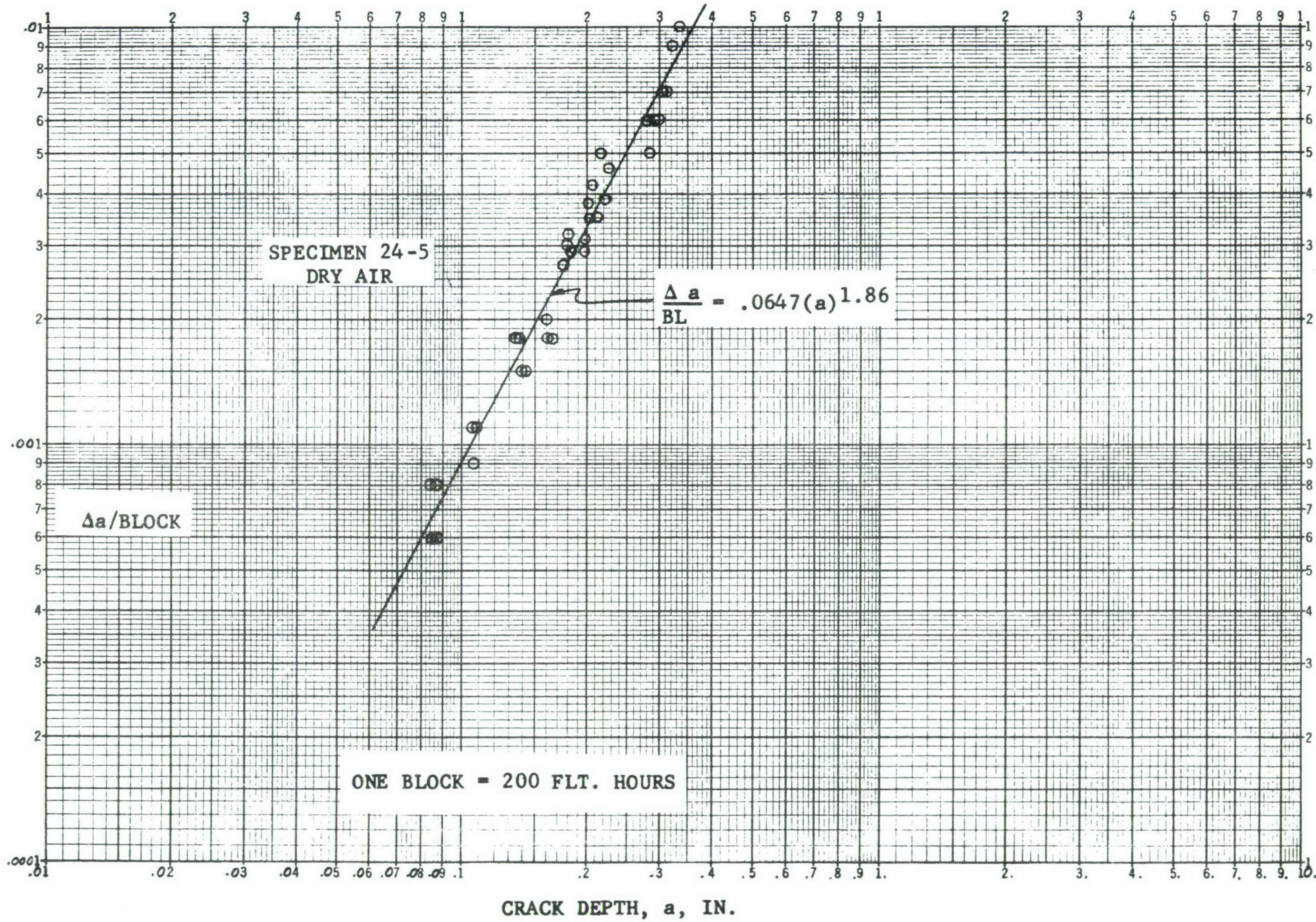


Figure 10 Baseline Wing Spectrum/Environmental Test Results
2024-T851 Aluminum Alloy

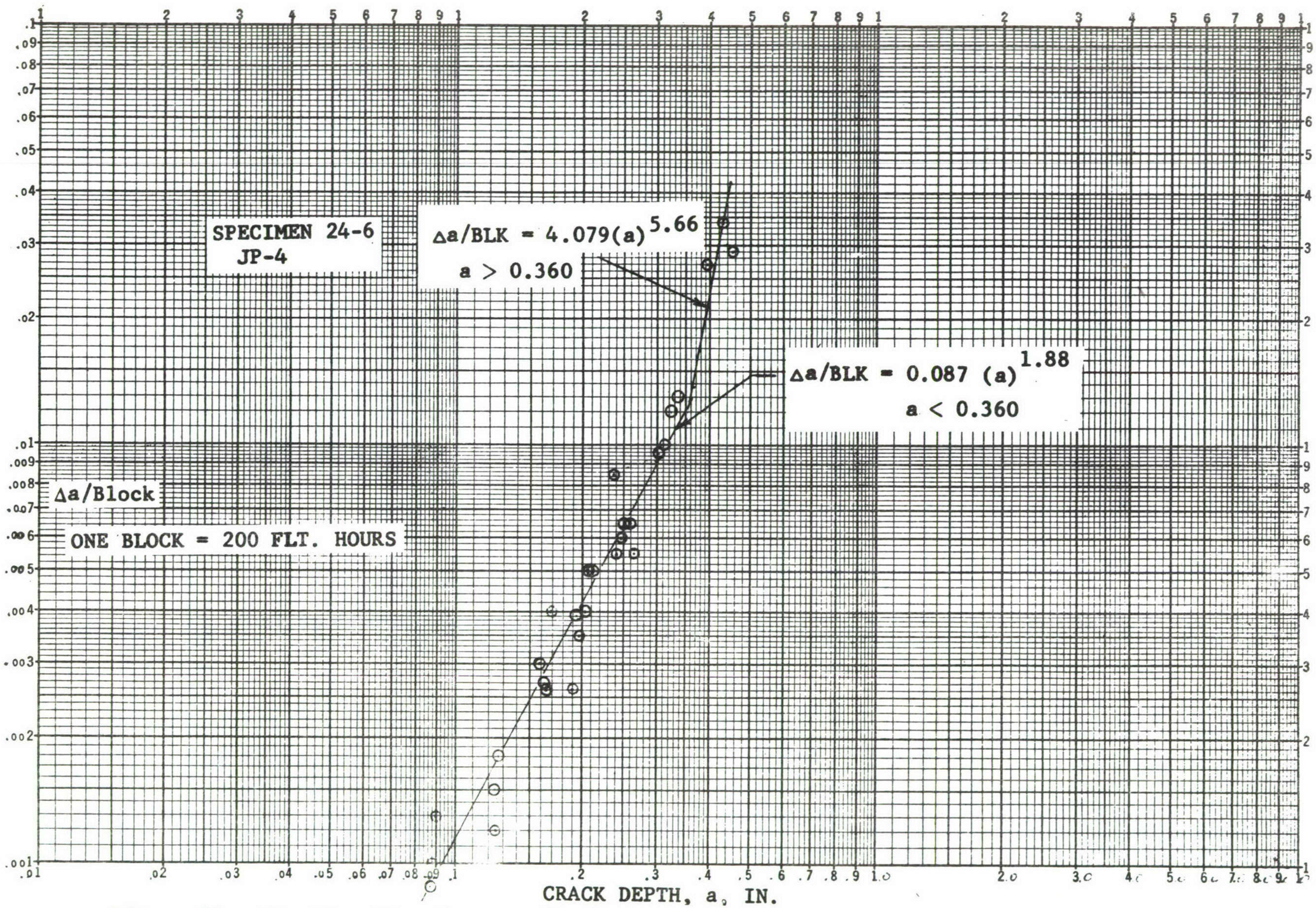


Figure 11 - Baseline Wing Spectrum/Environmental Test Results 2024-T851 Aluminum Alloy

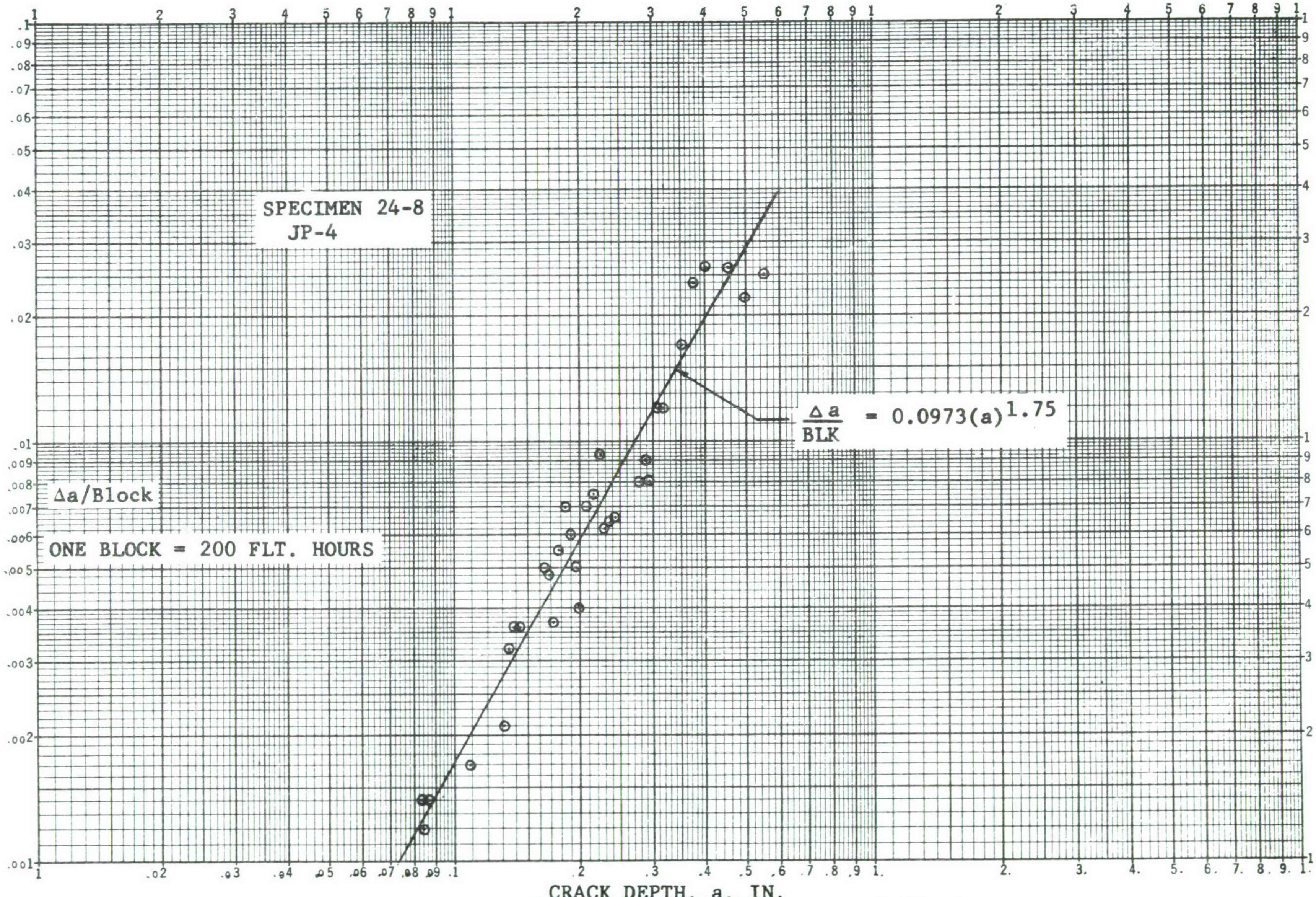


Figure 12 - Baseline Wing Spectrum/Environmental Test Results 2024-T851 Aluminum Alloy

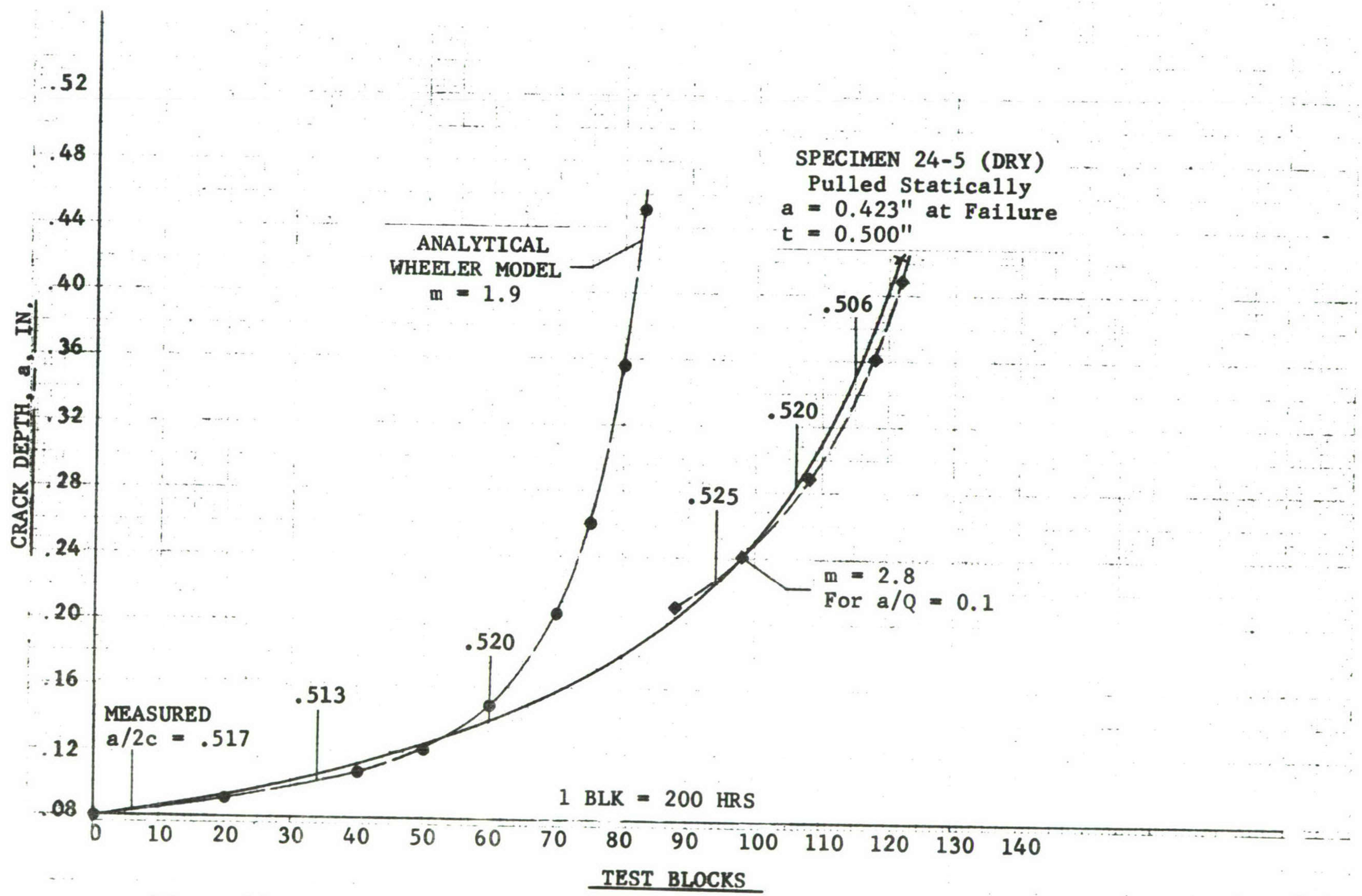


Figure 13 - Baseline Wing Spectrum/Environmental Test Results 2024-T851 Aluminum Alloy

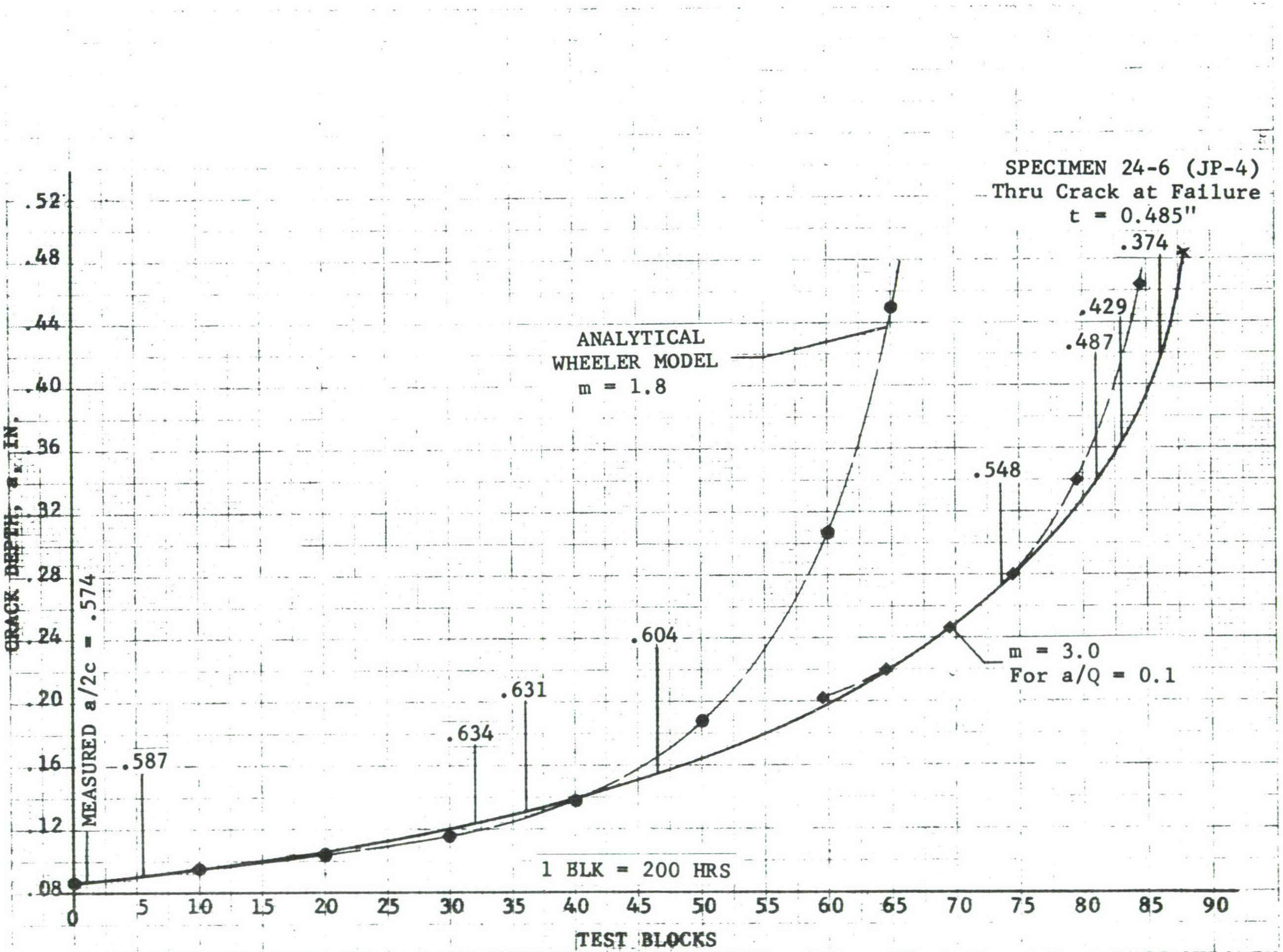


Figure 14 - Baseline Wing Spectrum/Environmental Test Results 2024-T851 Aluminum Alloy

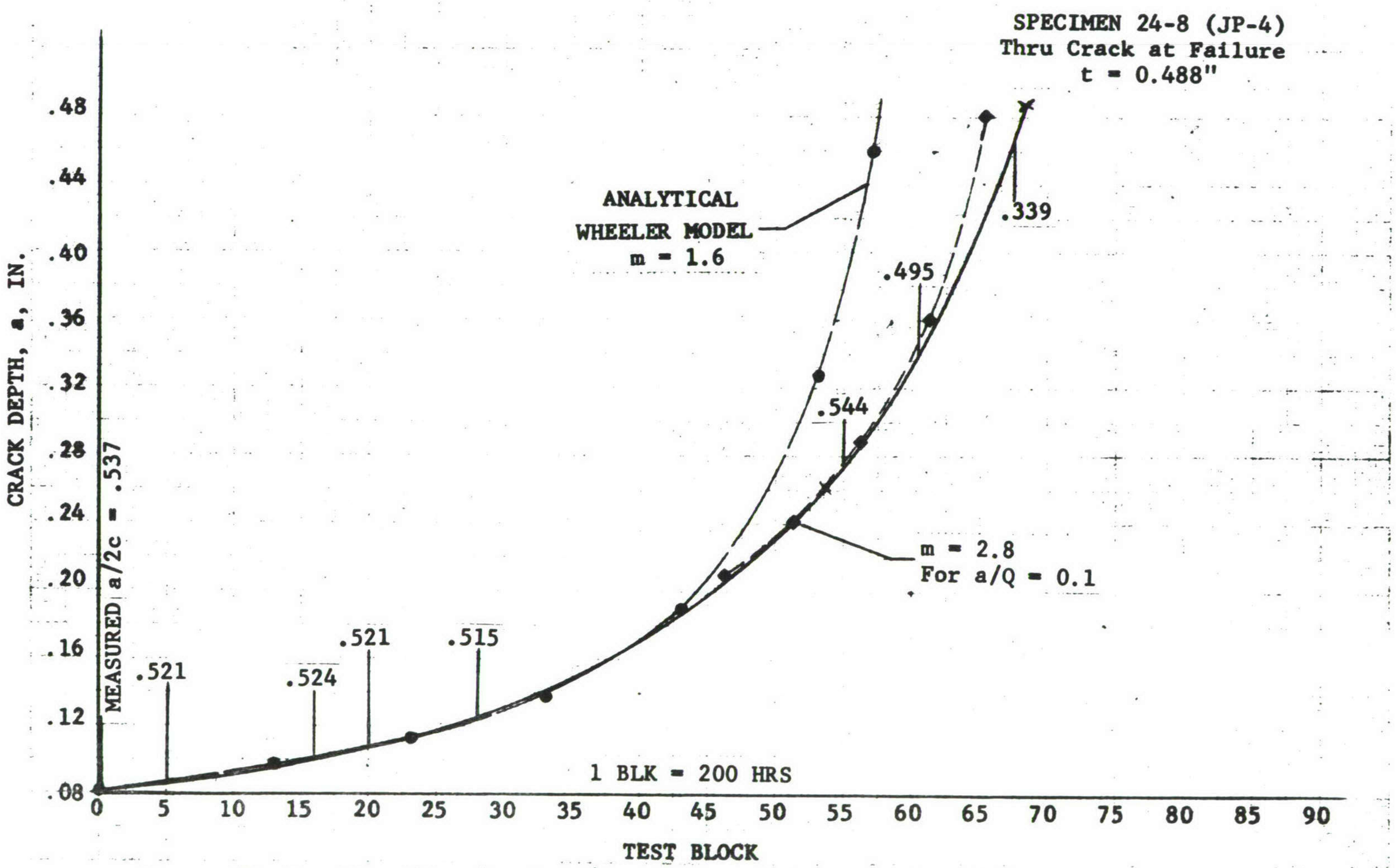


Figure 15 Baseline Wing Spectrum/Environmental Test Results 2024-T851 Aluminum Alloy

Table VII

ANALYSIS MATRIX

FOR FRACTURE DESIGN ALLOWABLE CURVES

FLAW TYPES	LOWER da/dN DATA			MID-POINT da/dN DATA			UPPER da/dN DATA		
	LOW K_{IC}	MID. K_{IC}	UPP. K_{IC}	LOW K_{IC}	MID. K_{IC}	UPP. K_{IC}	LOW K_{IC}	MID. K_{IC}	UPP. K_{IC}
o PART THROUGH SURFACE FLAWS									
o t = .25				X	X	X			
o t = .611	X	X	X	X	X	X	X	X	X
o t = .611 (Mild Usage)				X	X	X			
o t = 1.30				X	X	X			
o THROUGH THE THICKNESS BOLT HOLE FLAWS									
o 5/16 Dia.	X	X	X	X	X	X	X	X	X
o 5/16 Dia. (Mild Usage)				X	X	X			
o SEMI-CIRCULAR CORNER CRACK AT BOLT HOLES									
o 5/16 Dia.	X	X	X	X	X	X	X	X	X
o 5/16 Dia. (Mild Usage)				X	X	X			

NOTE: All analyses were performed using F-111 baseline severe usage spectrum (Phase I & II Training) except as noted in the Table.

Design allowable curves are presented in Figures 16 through 30 for each of the analyses indicated in Table VII except those involving mild usage. Mild usage is covered in paragraph IX.3.3. The flaw types, thicknesses, and bolt diameters are typical of those that might be assumed to occur in the baseline wing box structure as described previously. The following periods of unrepaired service usage are reflected in the allowable curves:

(1) 800 flight hours for a special visual inspection interval (one year operation times 2.0 = 400 x 2 = 800 hours)

(2) 2000 flight hours for a depot level inspection interval (1/4 lifetime times 2.0 = 1000 x 2 = 2000 hours)

(3) 4000 flight hours for non-inspectable failsafe multiple load path structure (one lifetime = 4000 hours)

(4) 8000 flight hours for non-inspectable slow crack growth structure (two lifetimes = 8000 hours)

The effect on life of constant design allowable stress levels is shown in Figures 31 through 39. This data was obtained by cross-plotting the design allowable curves in Figures 16 through 18 for the .611" thickness part through flaw, in Figures 25 through 27 for the through thickness bolt hole flaw, and in Figures 28 through 30 for the corner crack bolt hole flaw.

IX.3.1.4 Baseline Structural Weight Variations

The design allowable stress levels presented in the preceding section and elsewhere in this report generally require decreasing current baseline stress levels. The result is a delta weight penalty.

Stress level reduction required to meet damage tolerance requirements will be accomplished primarily by increasing lower skin thickness. No material changes are anticipated.

To quickly determine delta weight as a function of stress level, techniques developed for the analytical assembly design phase of the basic Convair Aerospace ADP Wing Contract were expanded for use in these studies.

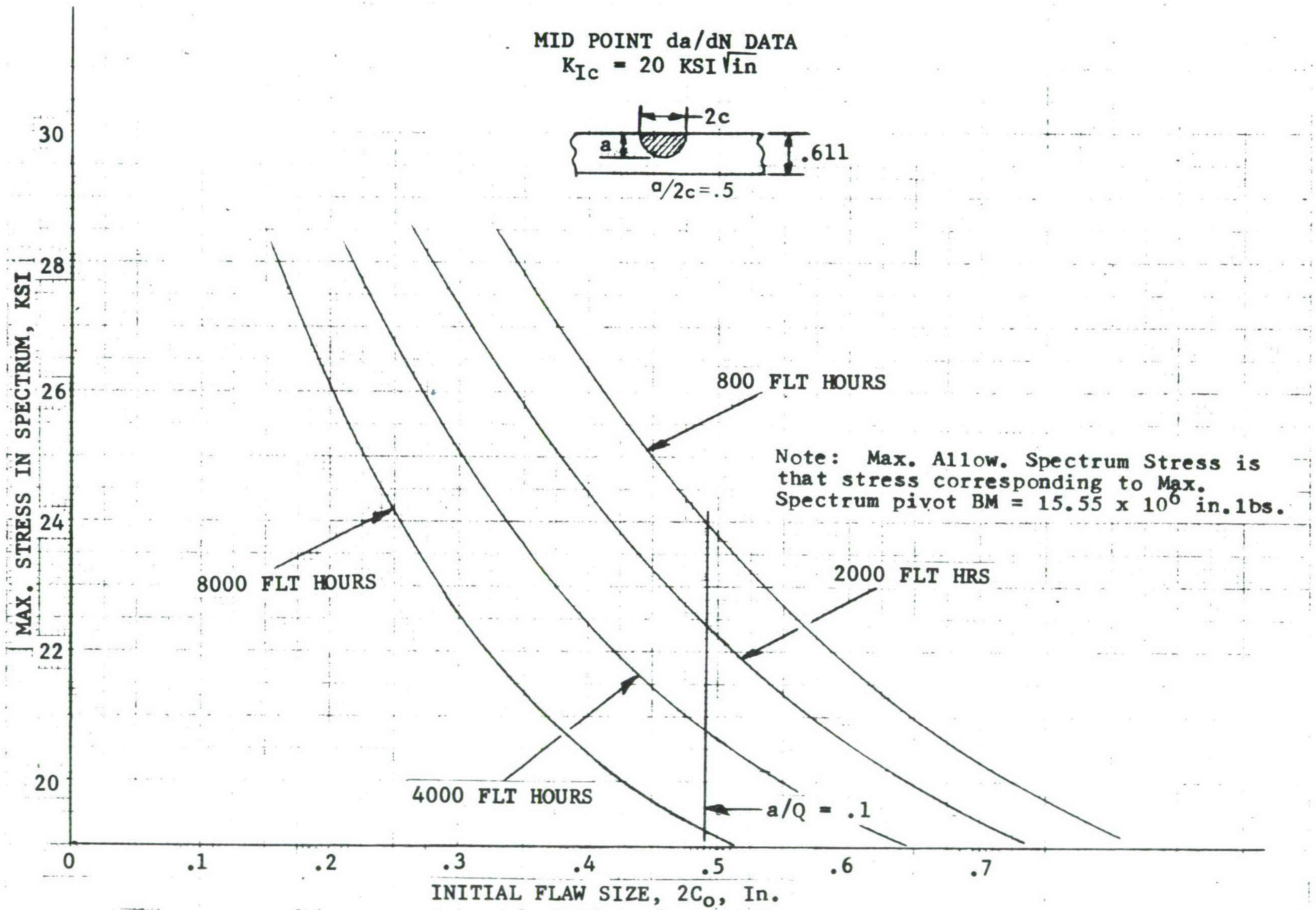


Figure 16 Allowable Curves for 2024-T851 Surface Flaws in .611 Skin

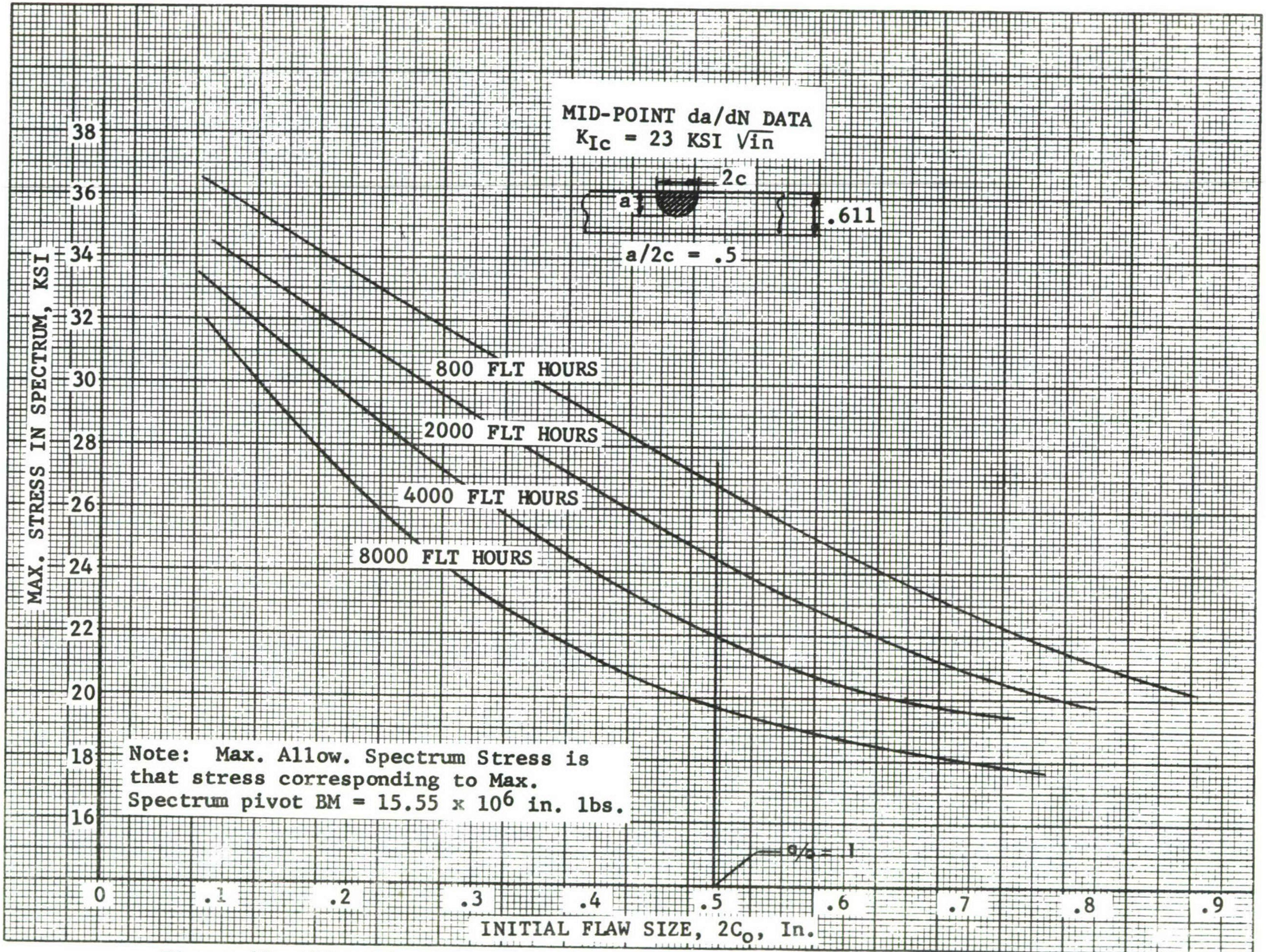


Figure 17 Allowable Curves for 2024-T851 Surface Flaws in .611 in. Skin

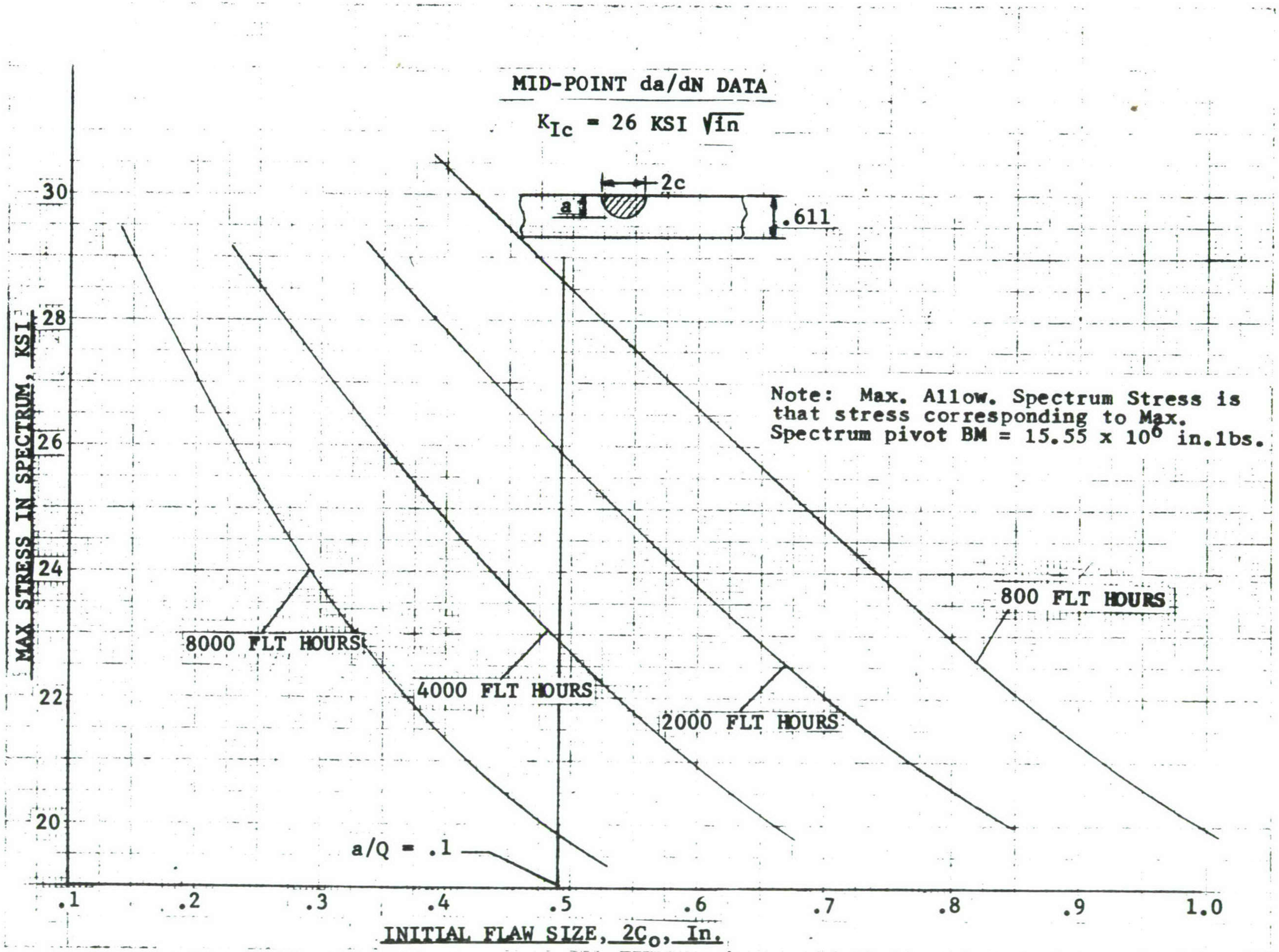


Figure 18 Allowable Curves for 2024-T851 Surface Flaws in .611 In. Skin .

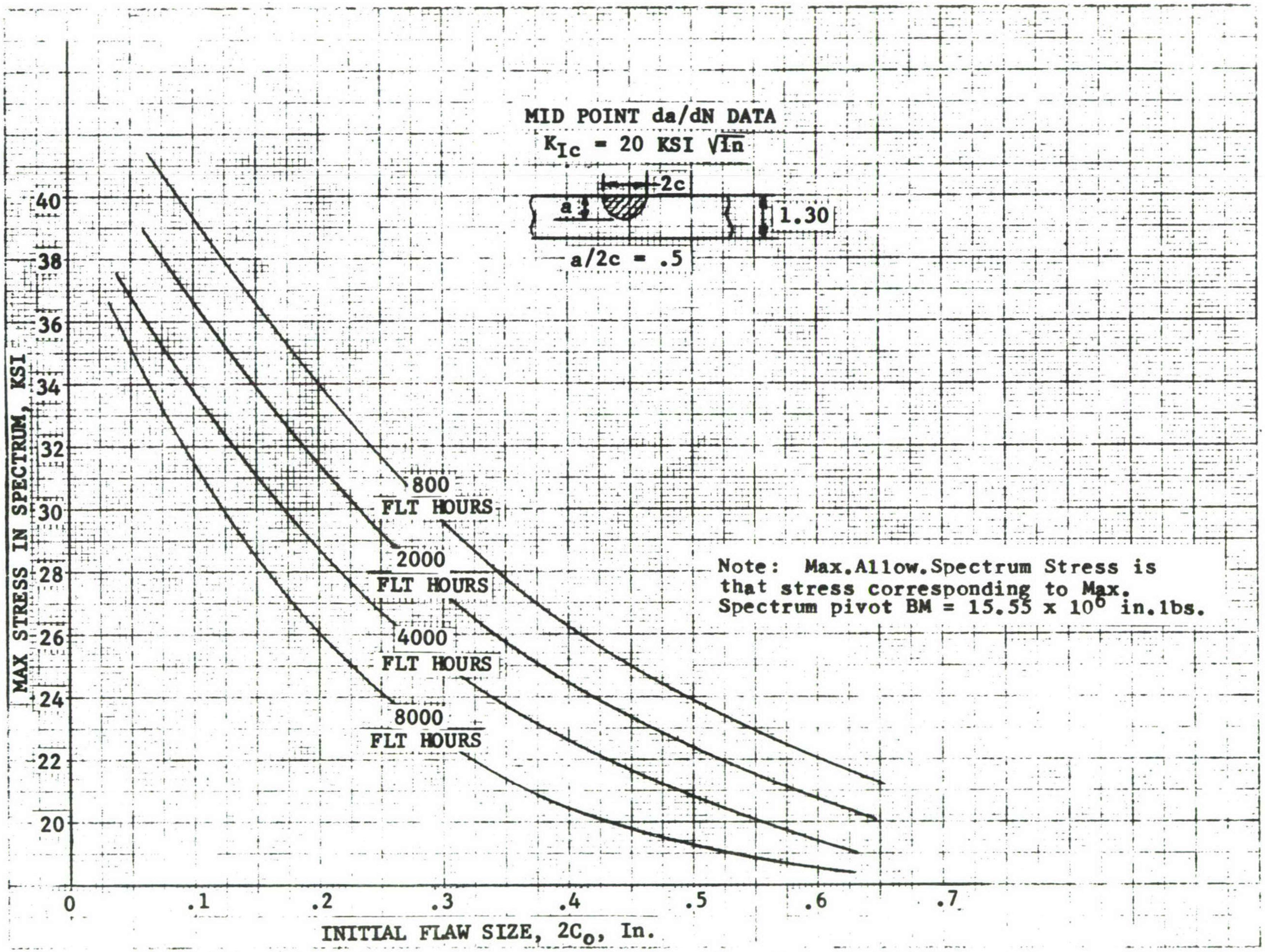


Figure 19 Allowable Curves for 2024-T851 Surface Flaws in 1.30 In. Skin

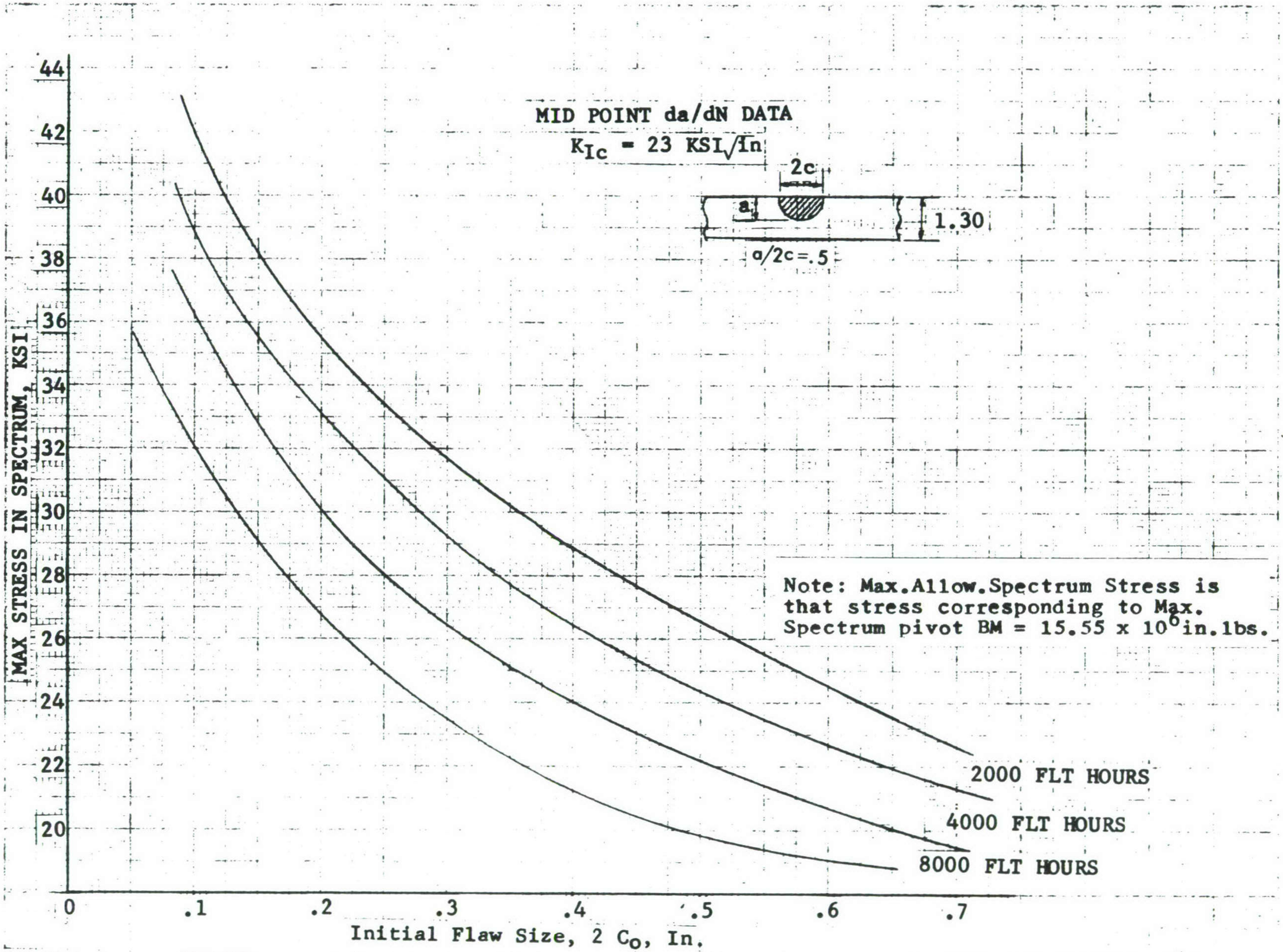


Figure 20 Allowable Curves for 2024-T851 Surface Flaws in 1.30 In. Skin

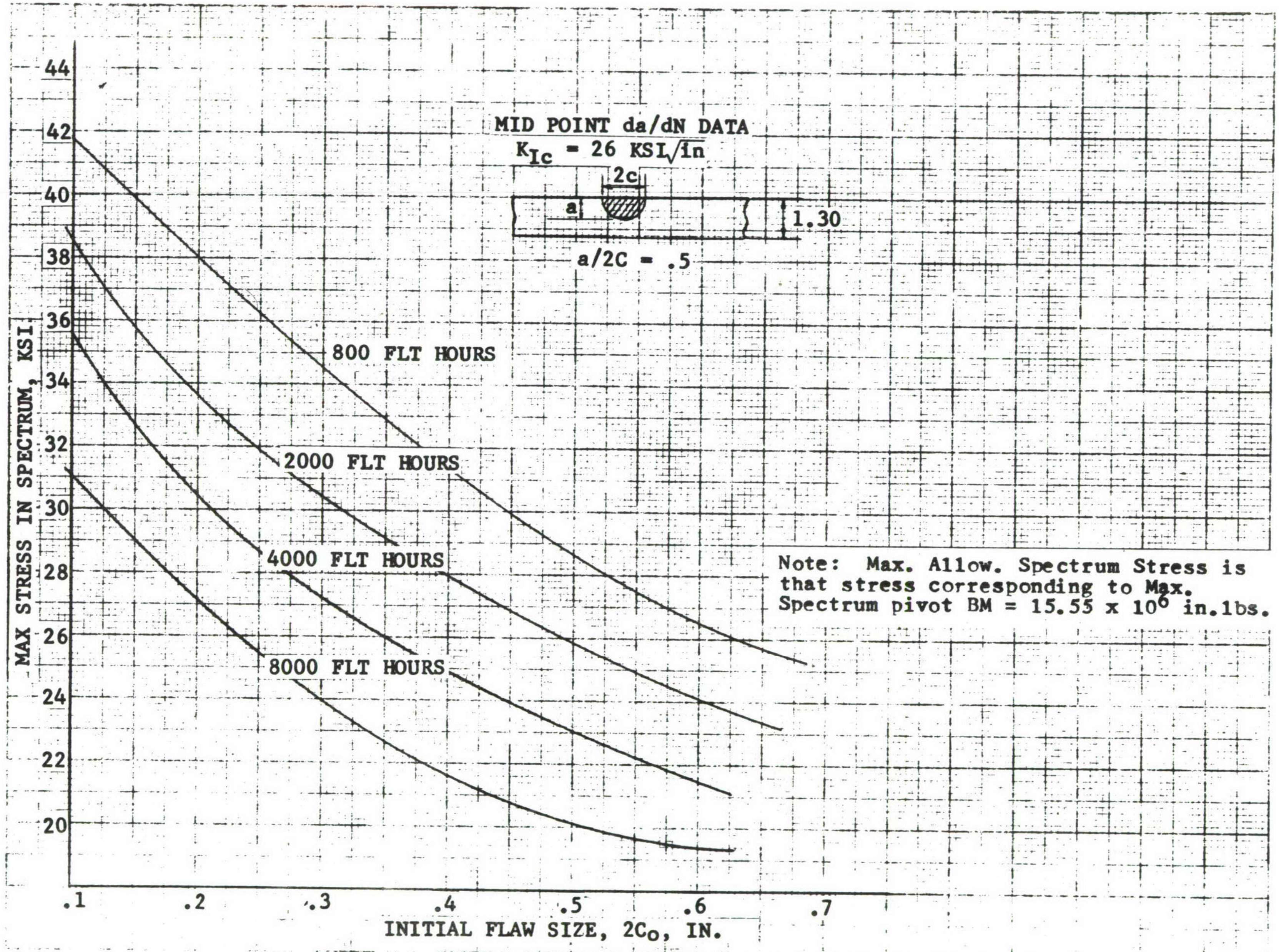


Figure 21. Allowable Curves for 2024-T851 Surface Flaws in 1.30 In. Skin

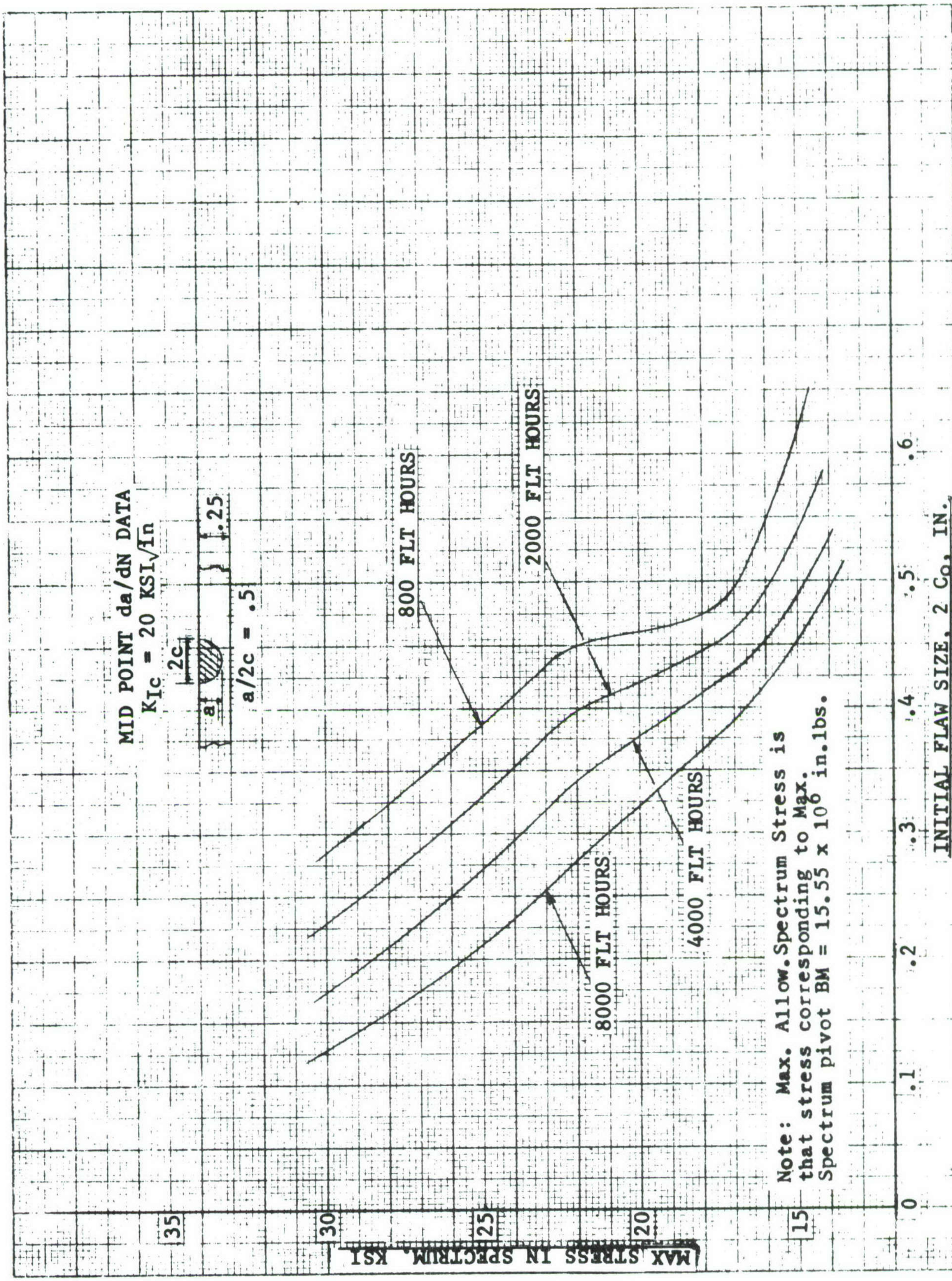


Figure 22 Allowable Curves for 2024-T851 Surface Flaws in .25 In. Spar Cap

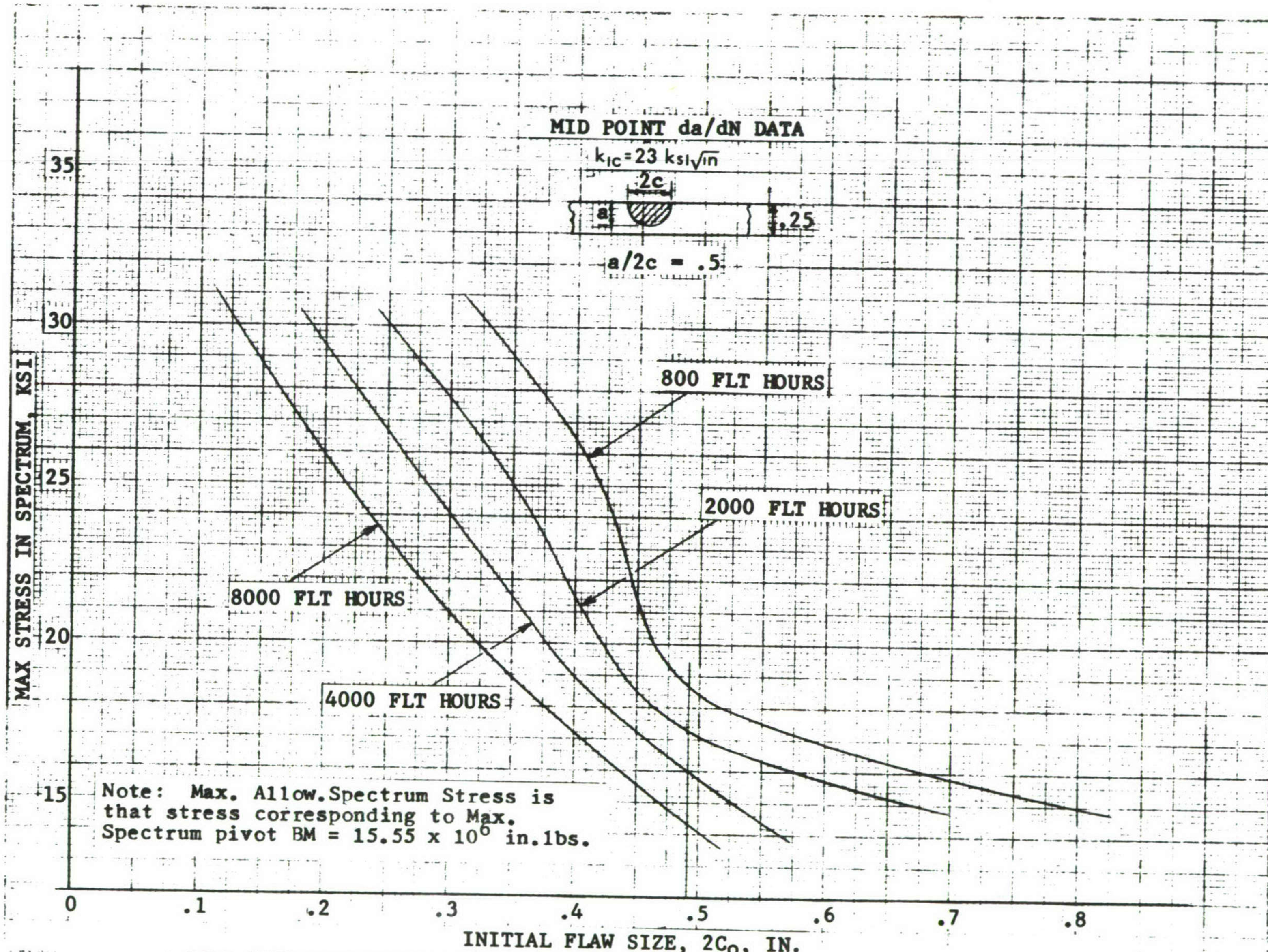


Figure 23 Allowable Curves for 2024-T851 Surface Flaws in .25 In. Spar Cap

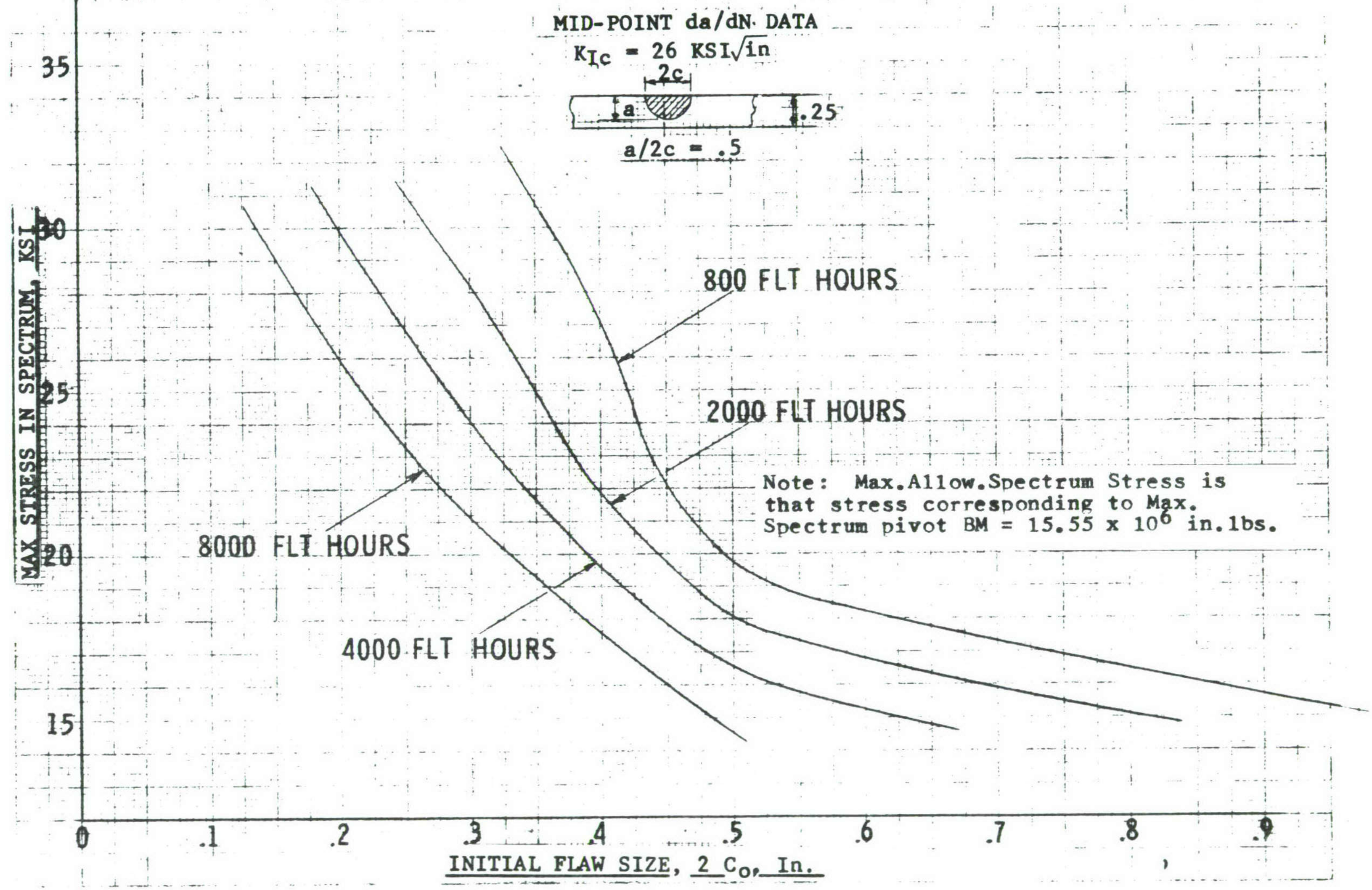


Figure 24 Allowable Curves for 2024-T851 Surface Flaws in .25 In. Spar Cap

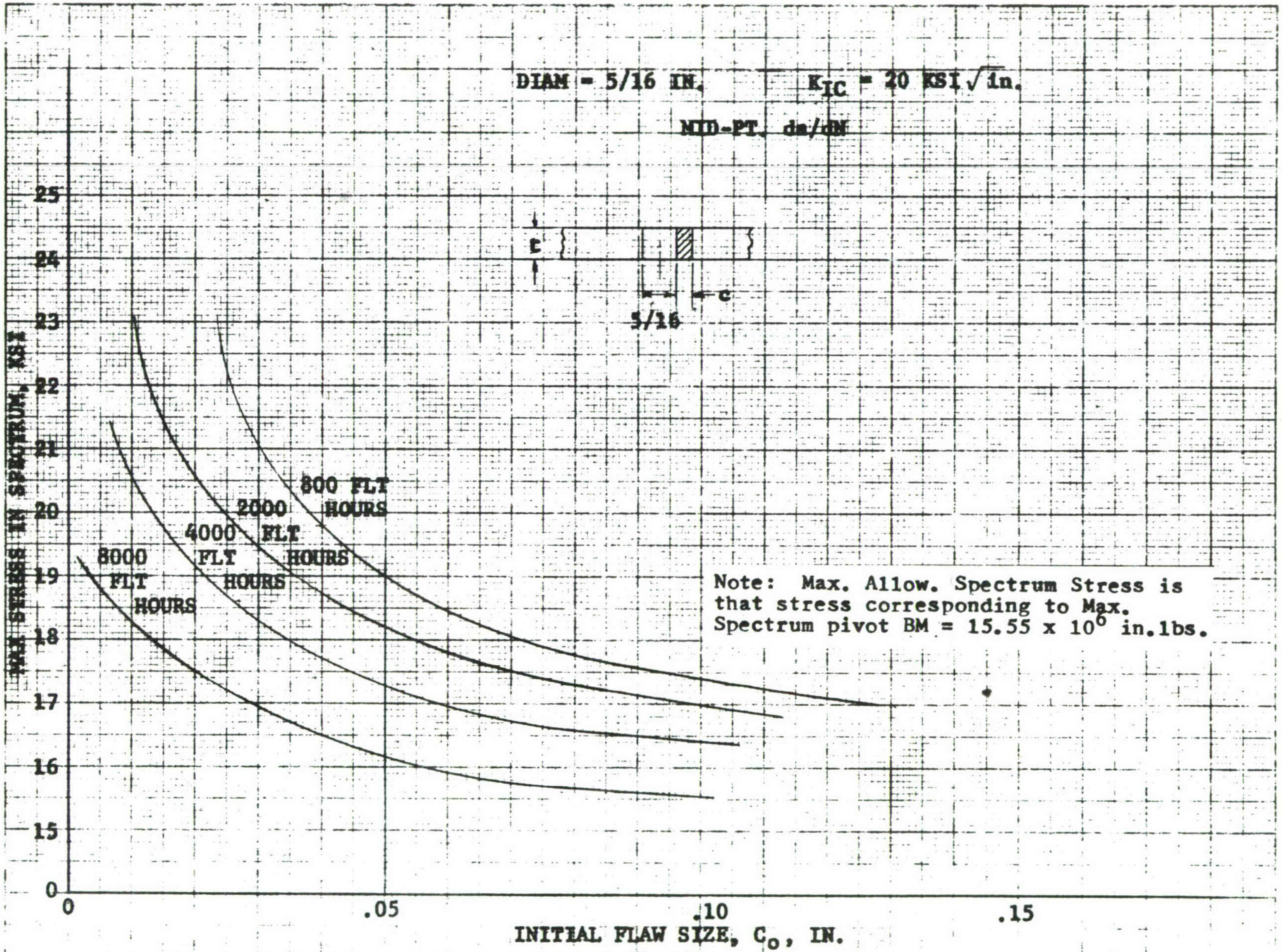


Figure 25 Allowable Curves for 2024-T851 Through Flaws at a Bolt Hole

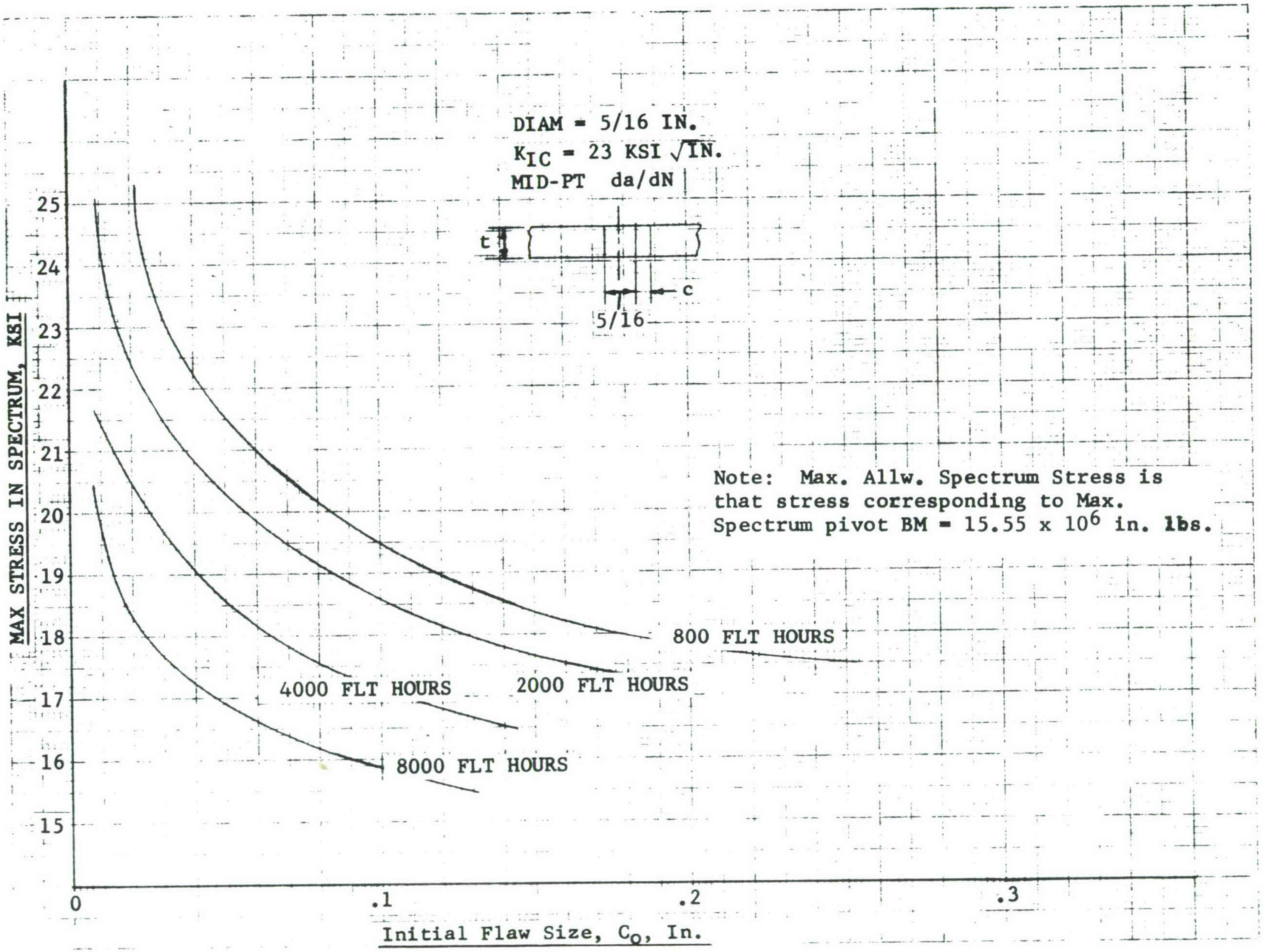


Figure 26 Allowable Curves for 2024-T851 Through Flaws at a Bolt Hole

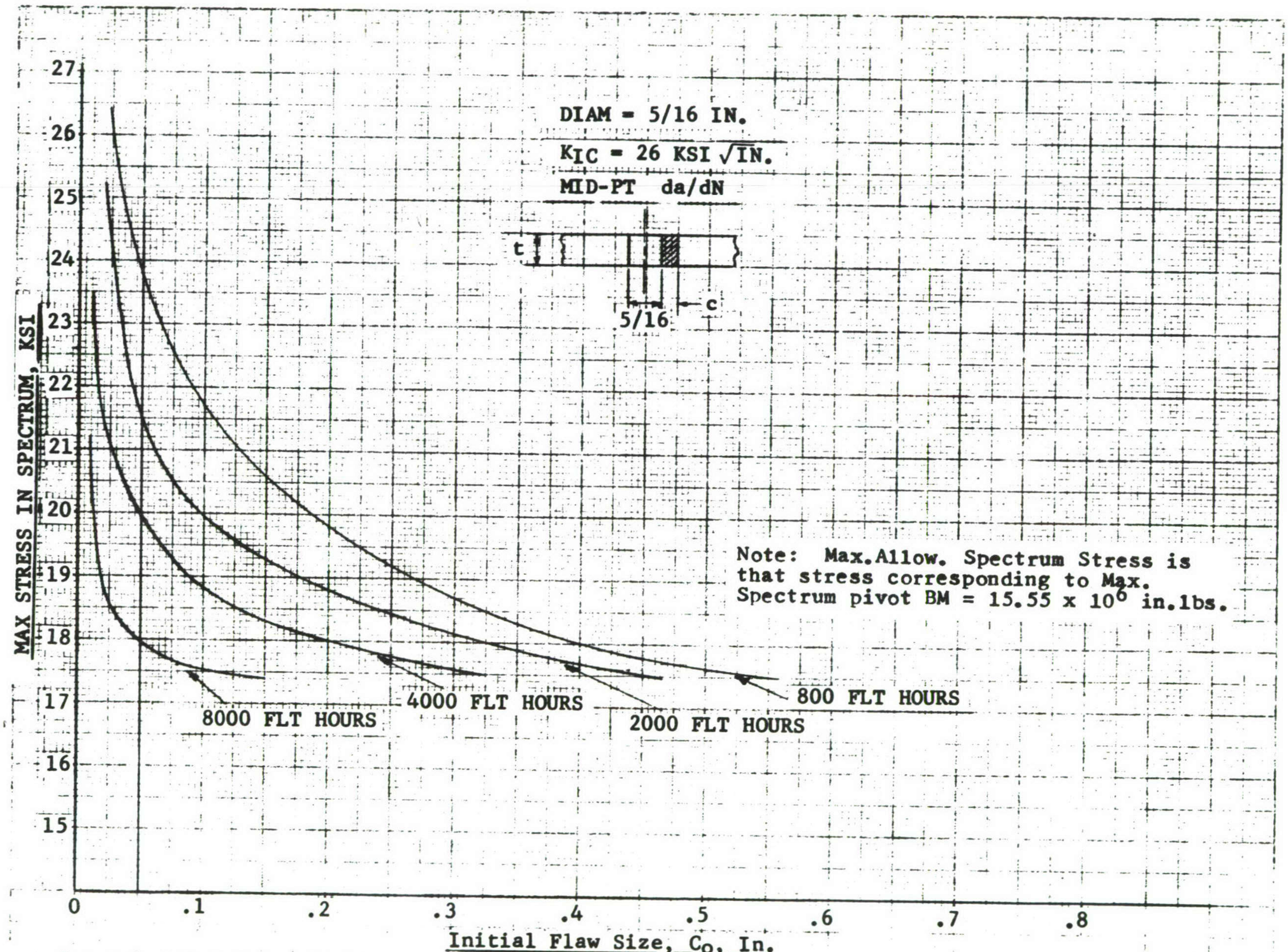


Figure 27 Allowable Curves for 2024-T851 Through Flaws at a Bolt Hole

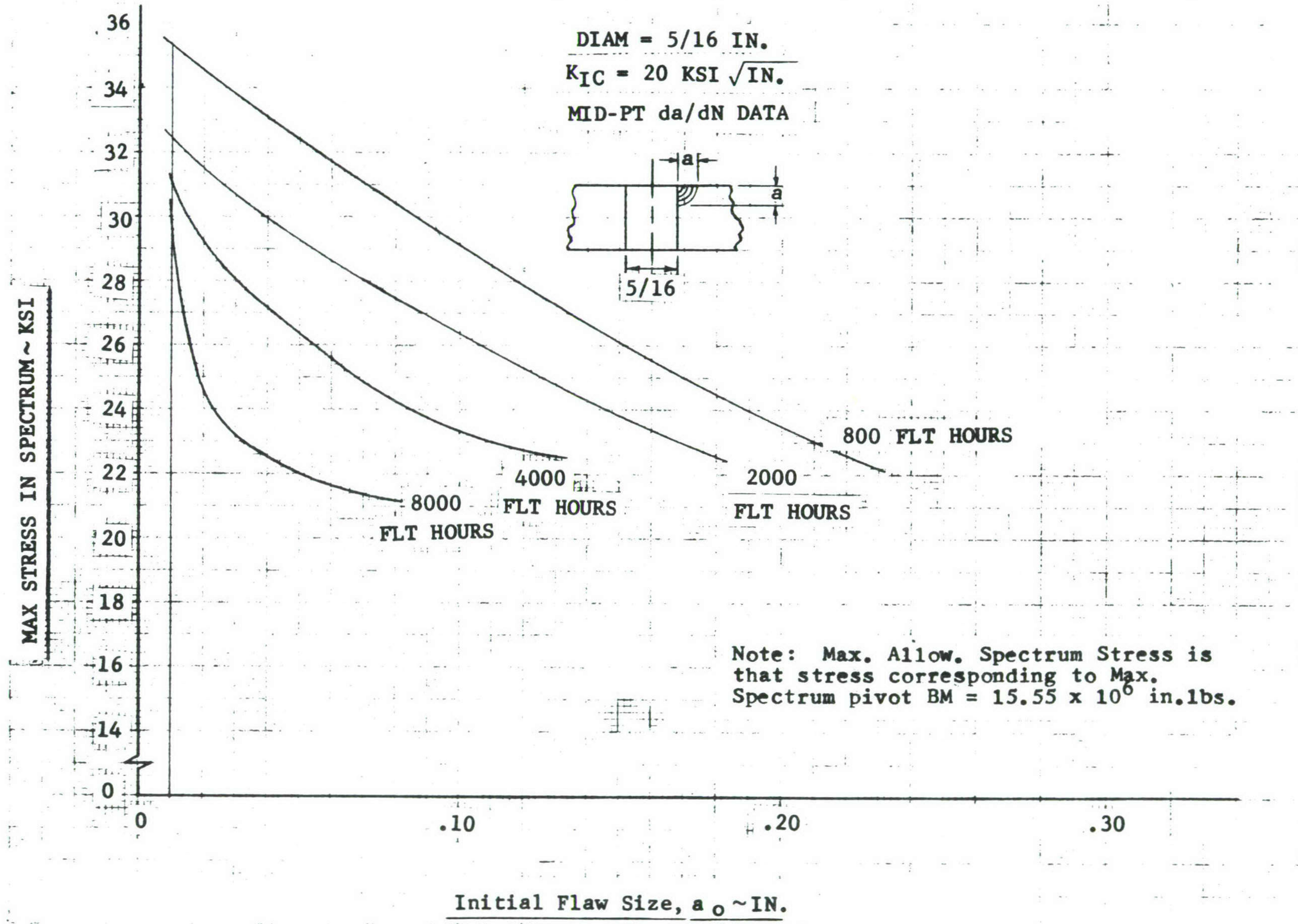


Figure 28 Allowable Curves for 2024-T851 Semi-Circular Corner Crack at a Hole

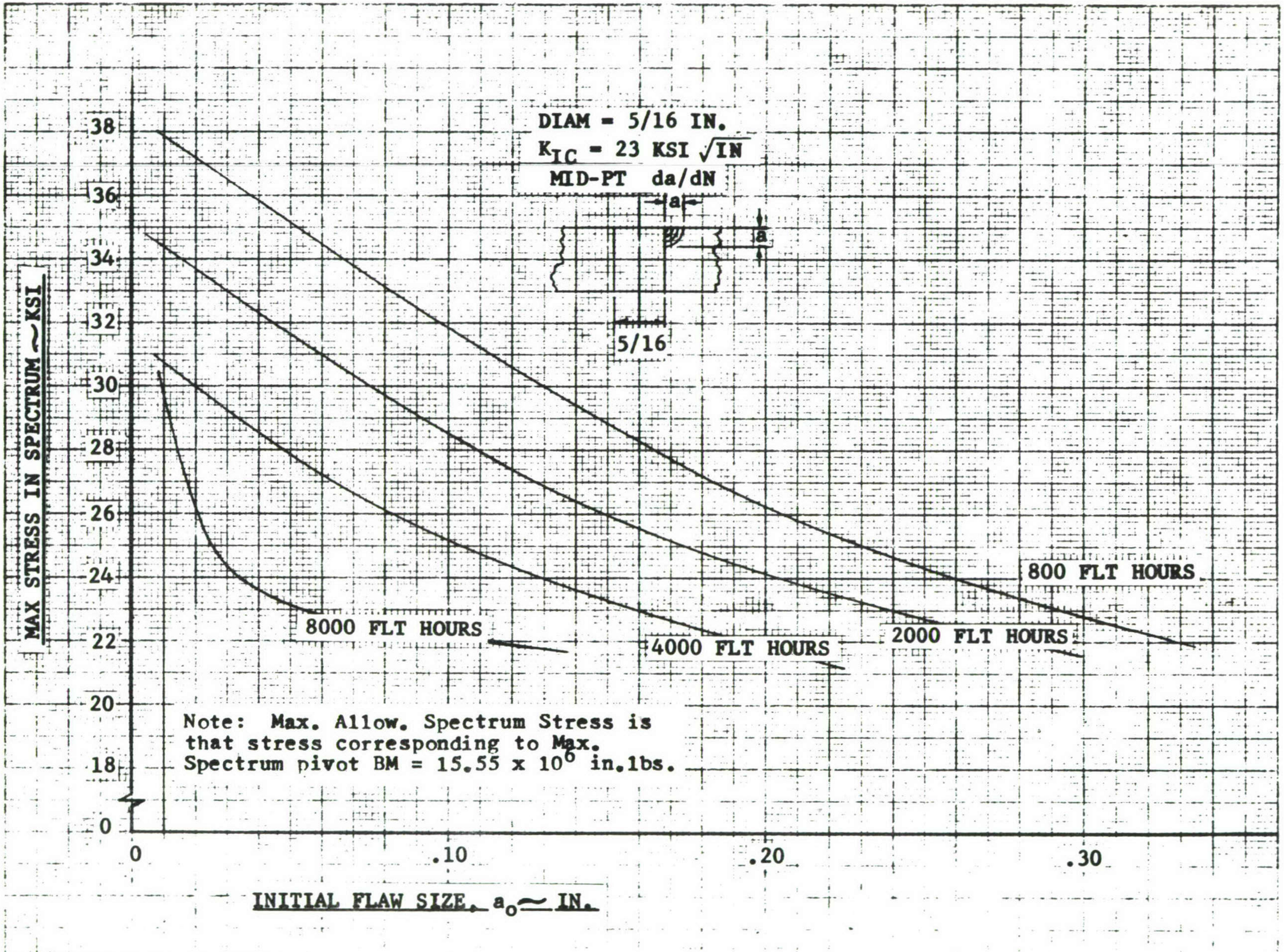


Figure 29 Allowable Curves for 2024-T851 Semi-Circular Corner Crack at a Hole

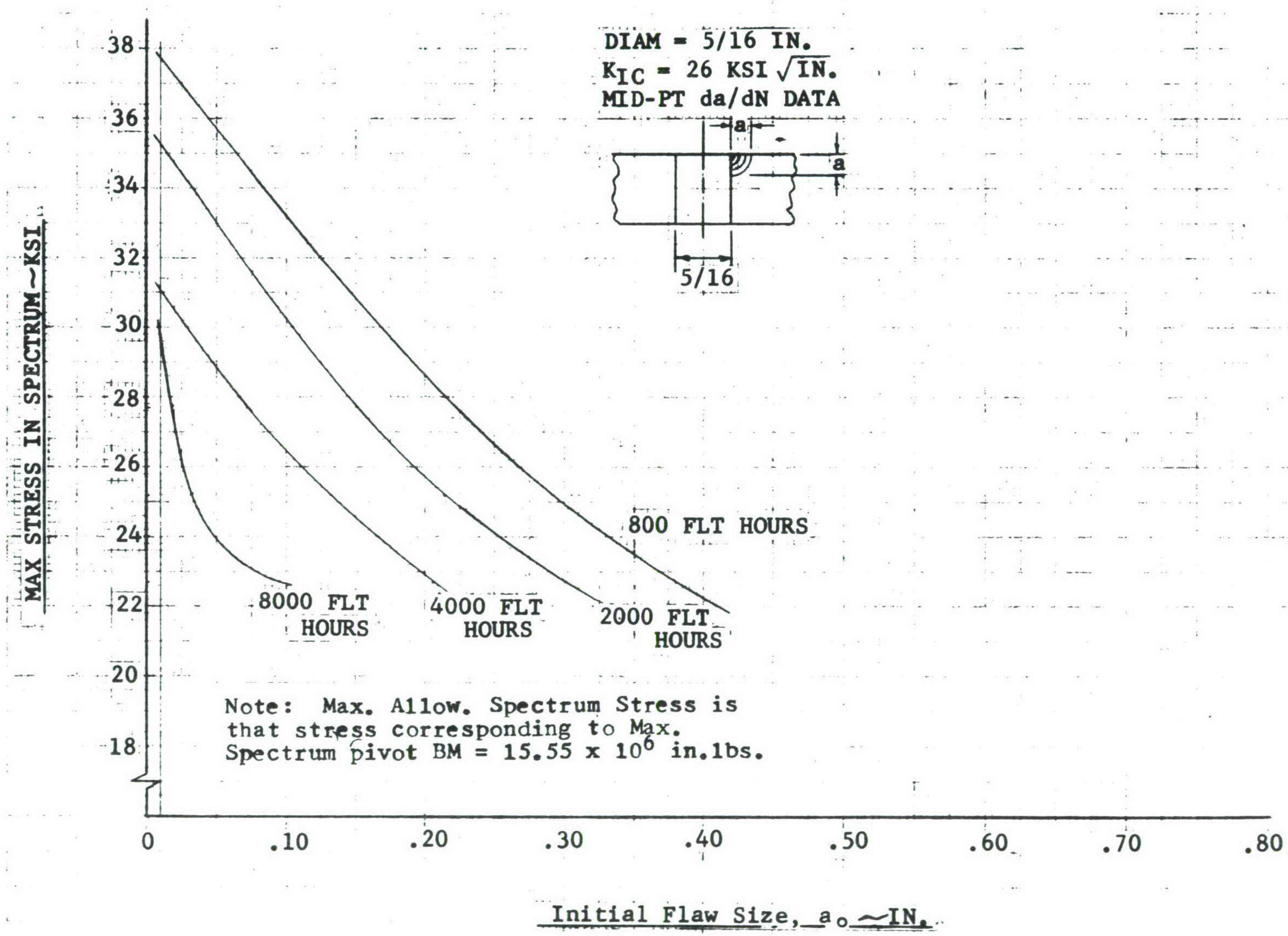


Figure 30 Allowable Curves for 2024-T851 Semi-Circular Corner Crack at a Hole

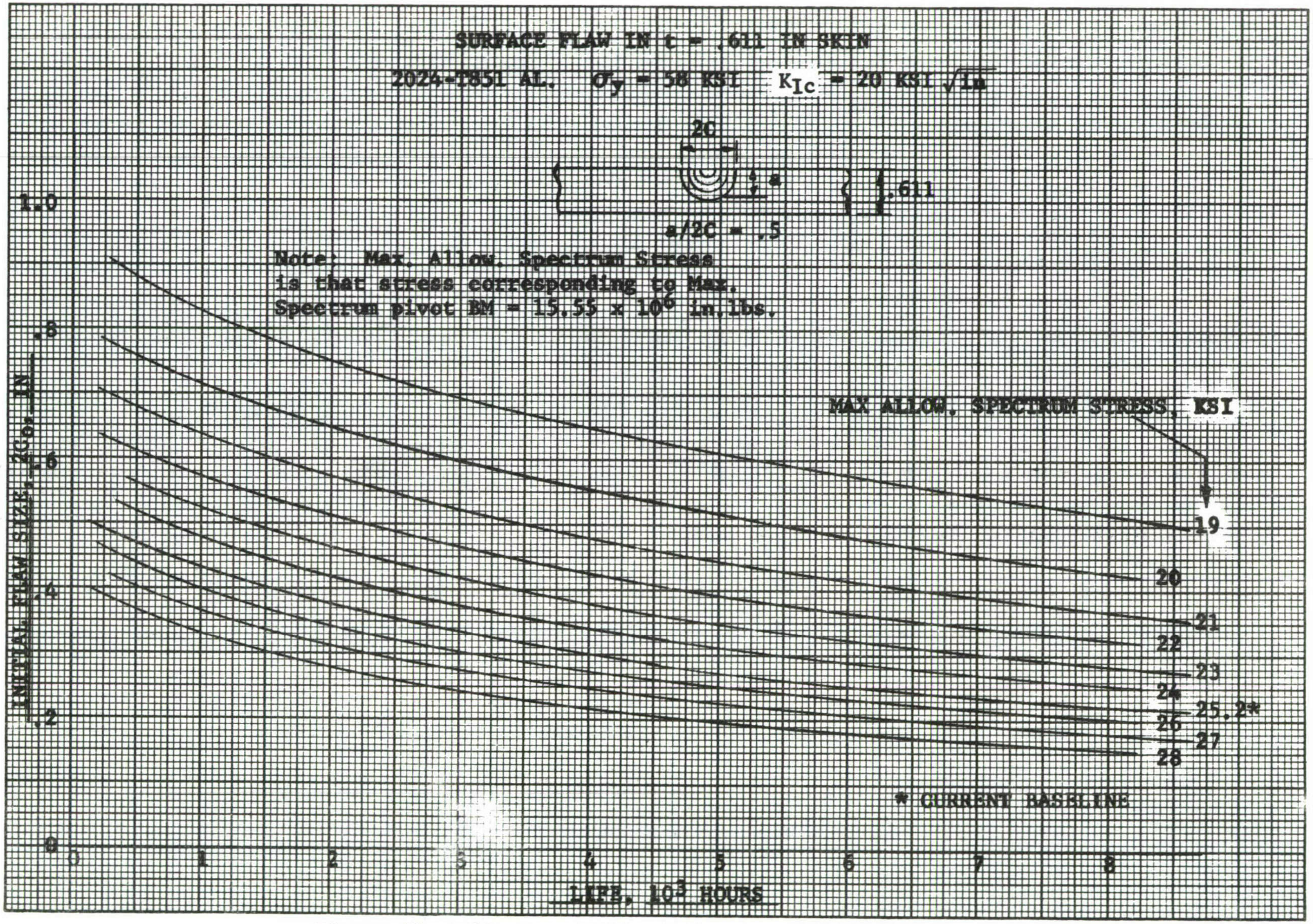


Figure 31 Effect on Life of Constant Allowable Stress

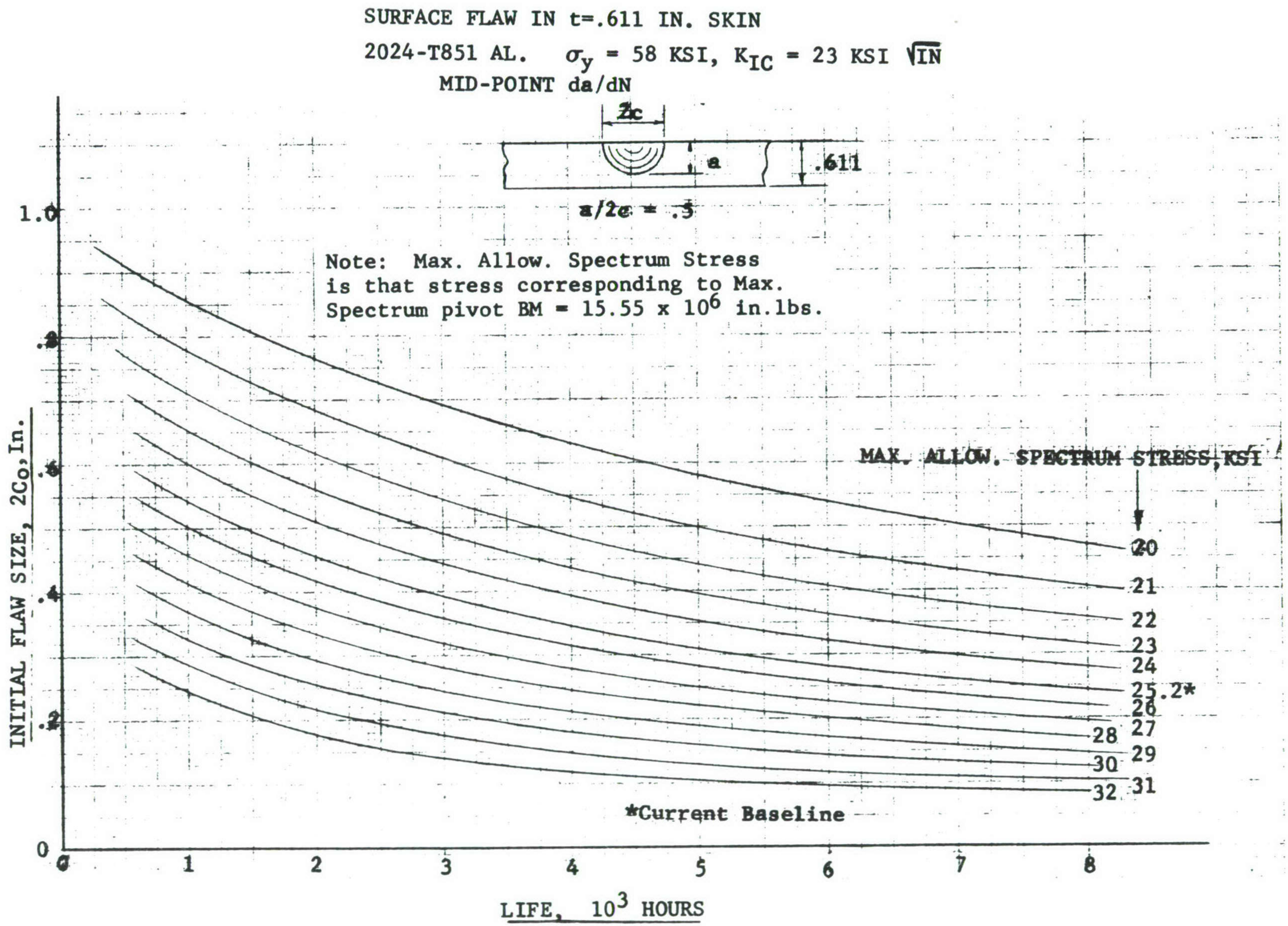


Figure 32 Effect On Life Of Constant Allowable Stress

SURFACE FLAW IN t = .611 IN SKIN

2024-T851 AL. σ_y = 58 KSI K_{IC} = 26 KSI $\sqrt{IN}$

MID-POINT da/dN

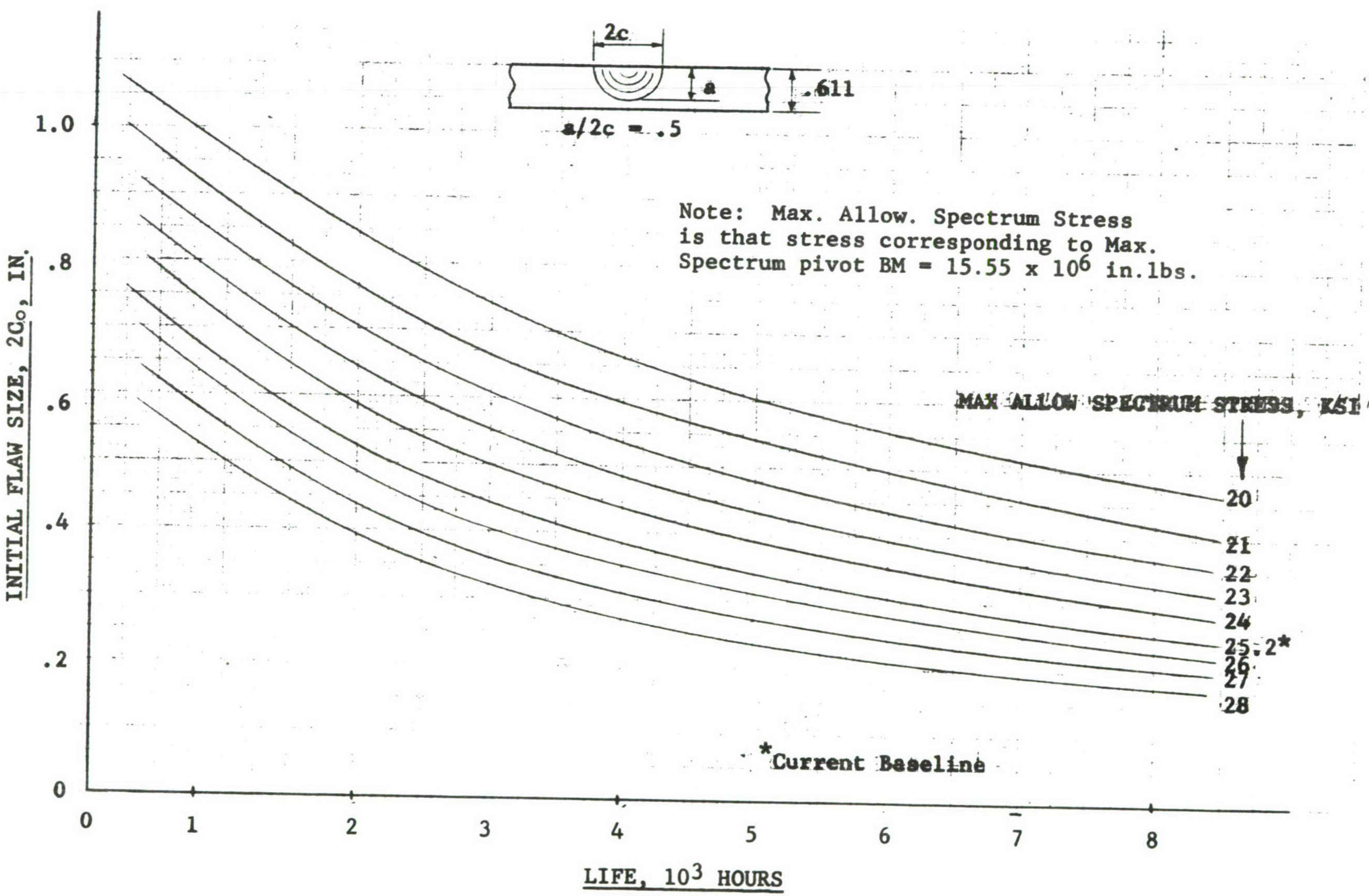


Figure 33 Effect on Life of Constant Allowable Stress

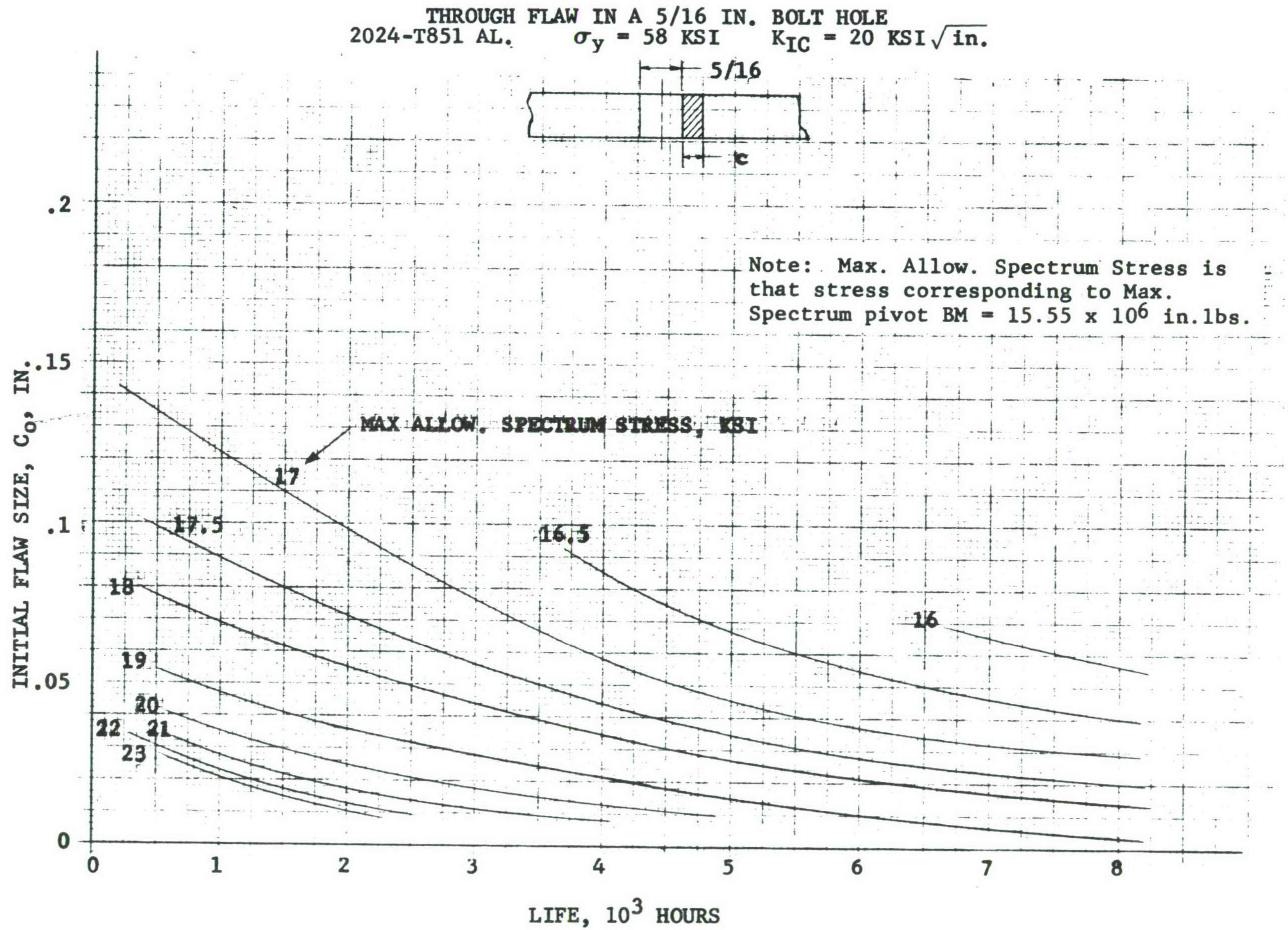


Figure 34 Effect on Life of Constant Allowable Stress

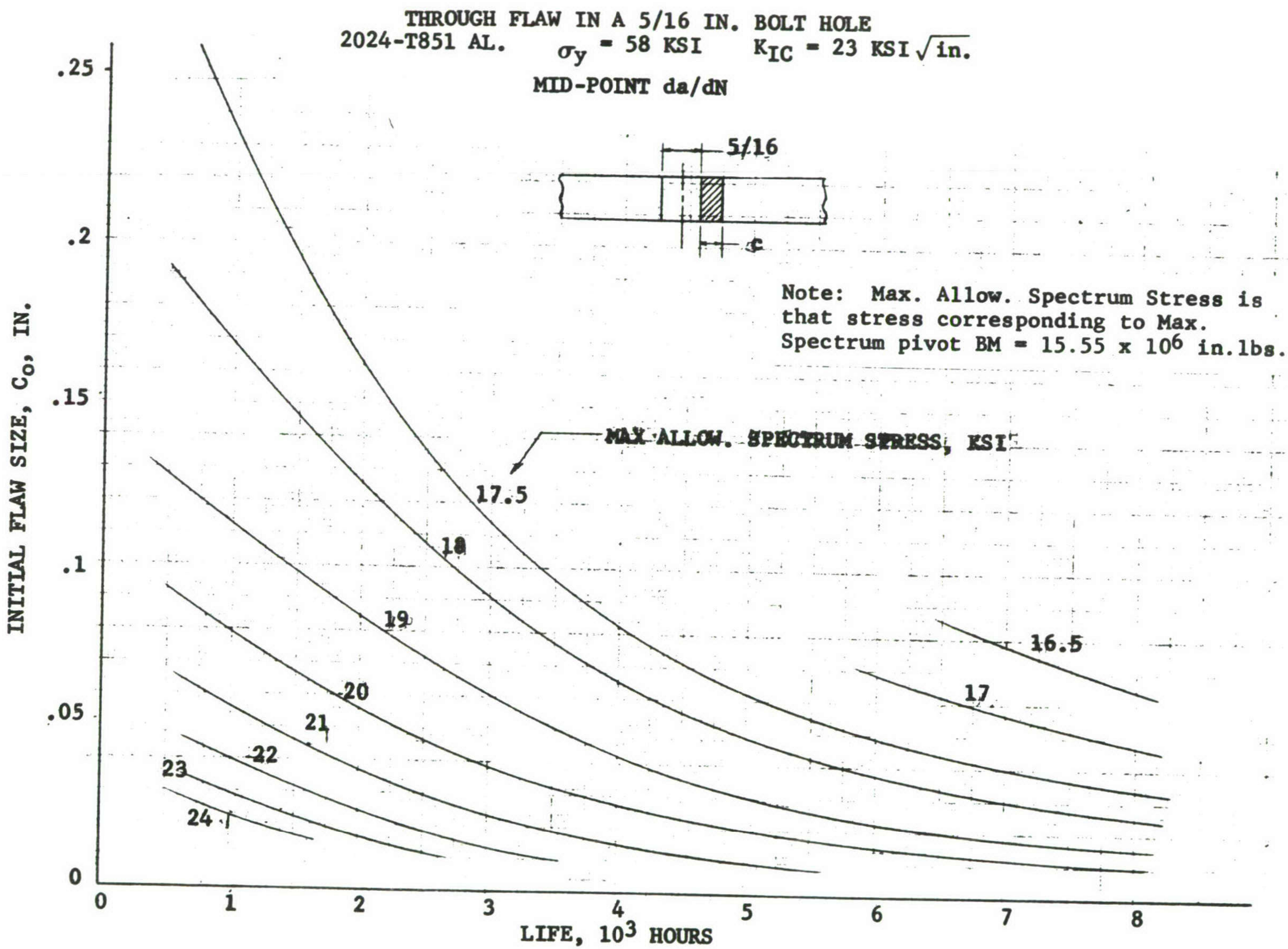


Figure 35 Effect on Life of Constant Allowable Stress

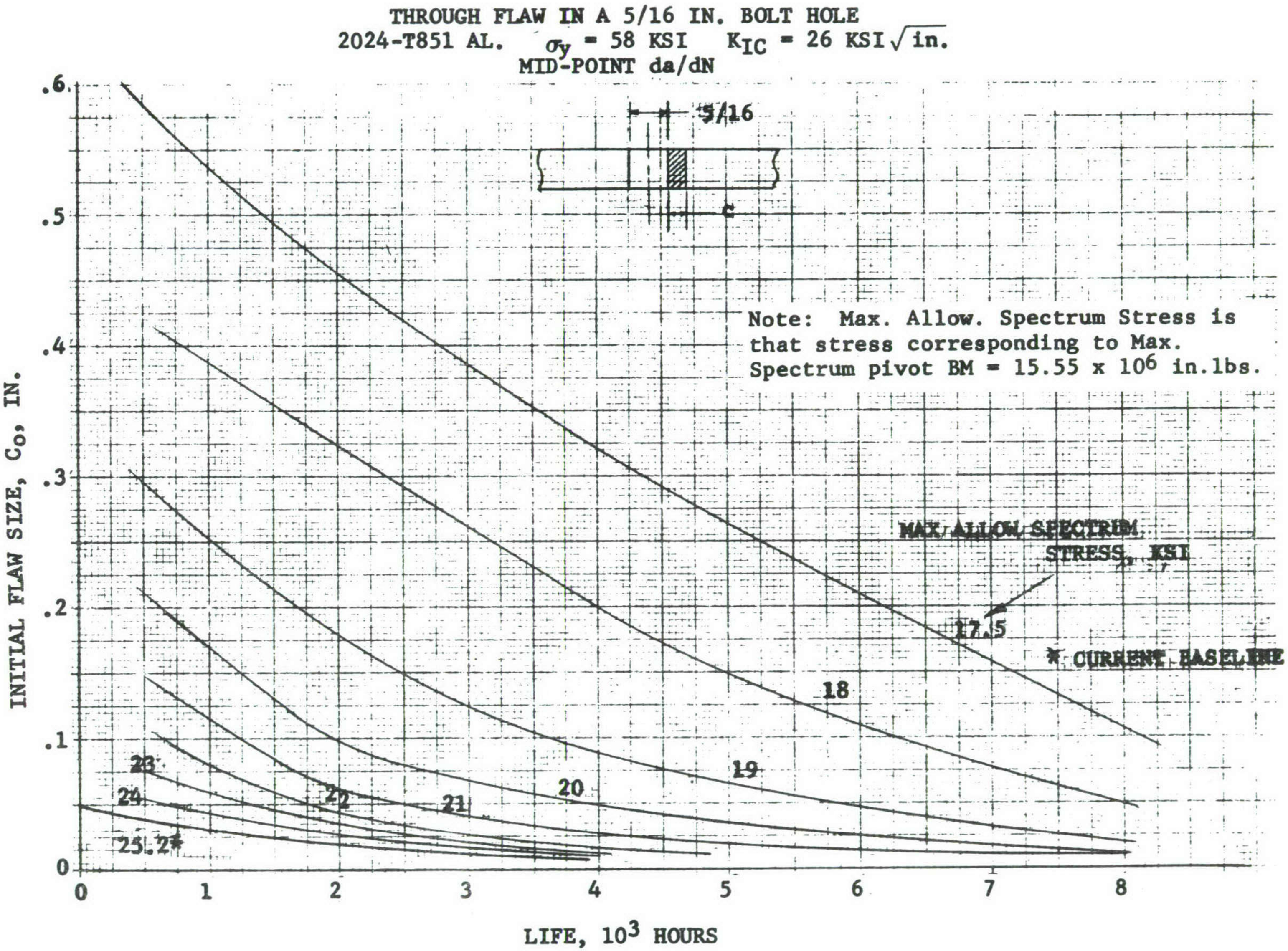


Figure 36 Effect on Life of Constant Allowable Stress

CORNER CRACK IN A 5/16 IN. BOLT HOLE
2024-T851 AL. σ_y = 58 KSI K_{IC} = 20 KSI$\sqrt{\text{in.}}$

MID-POINT da/dN

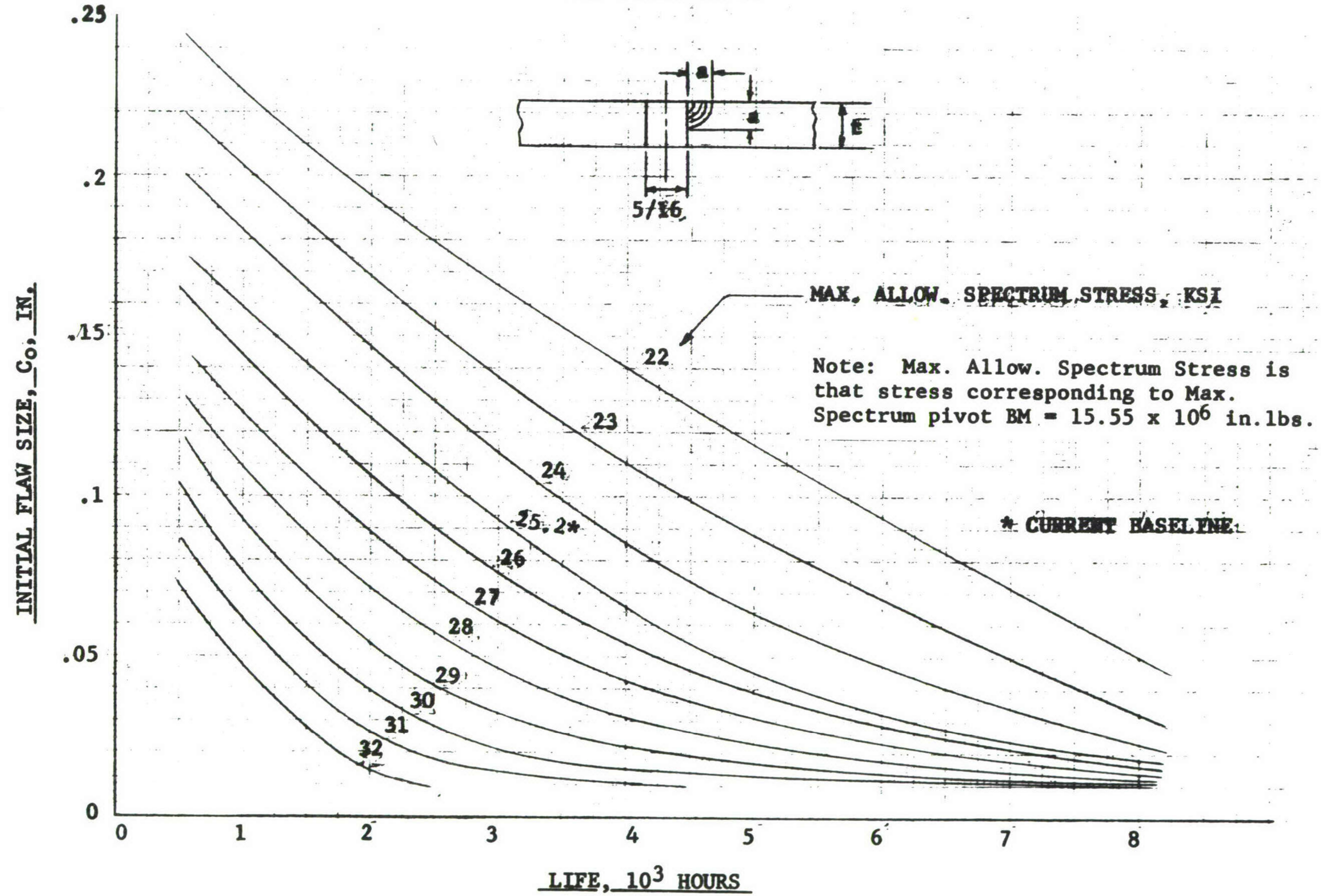


Figure 37 Effect on Life of Constant Allowable Stress

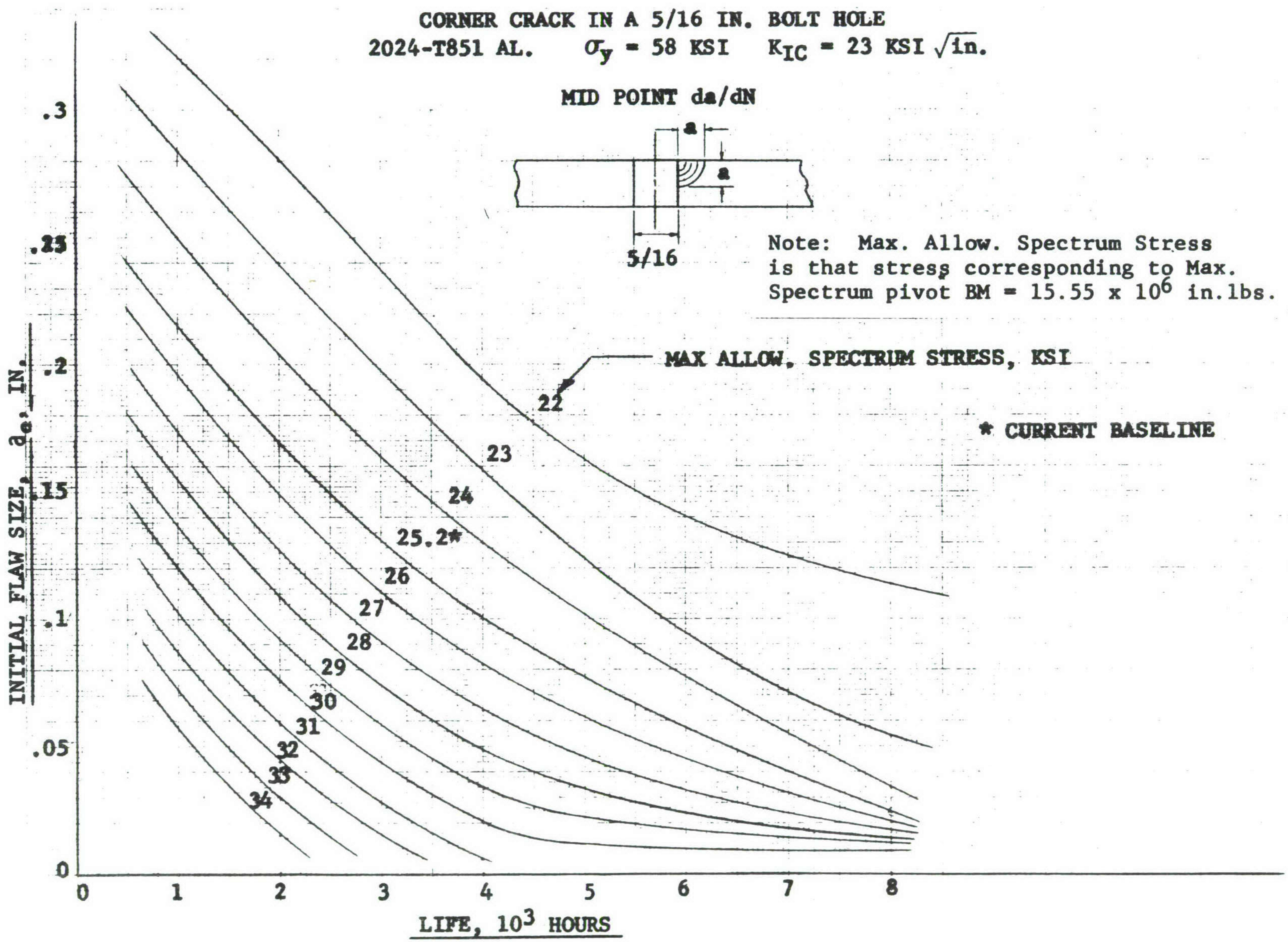


Figure 38 Effect on Life of Constant Allowable Stress

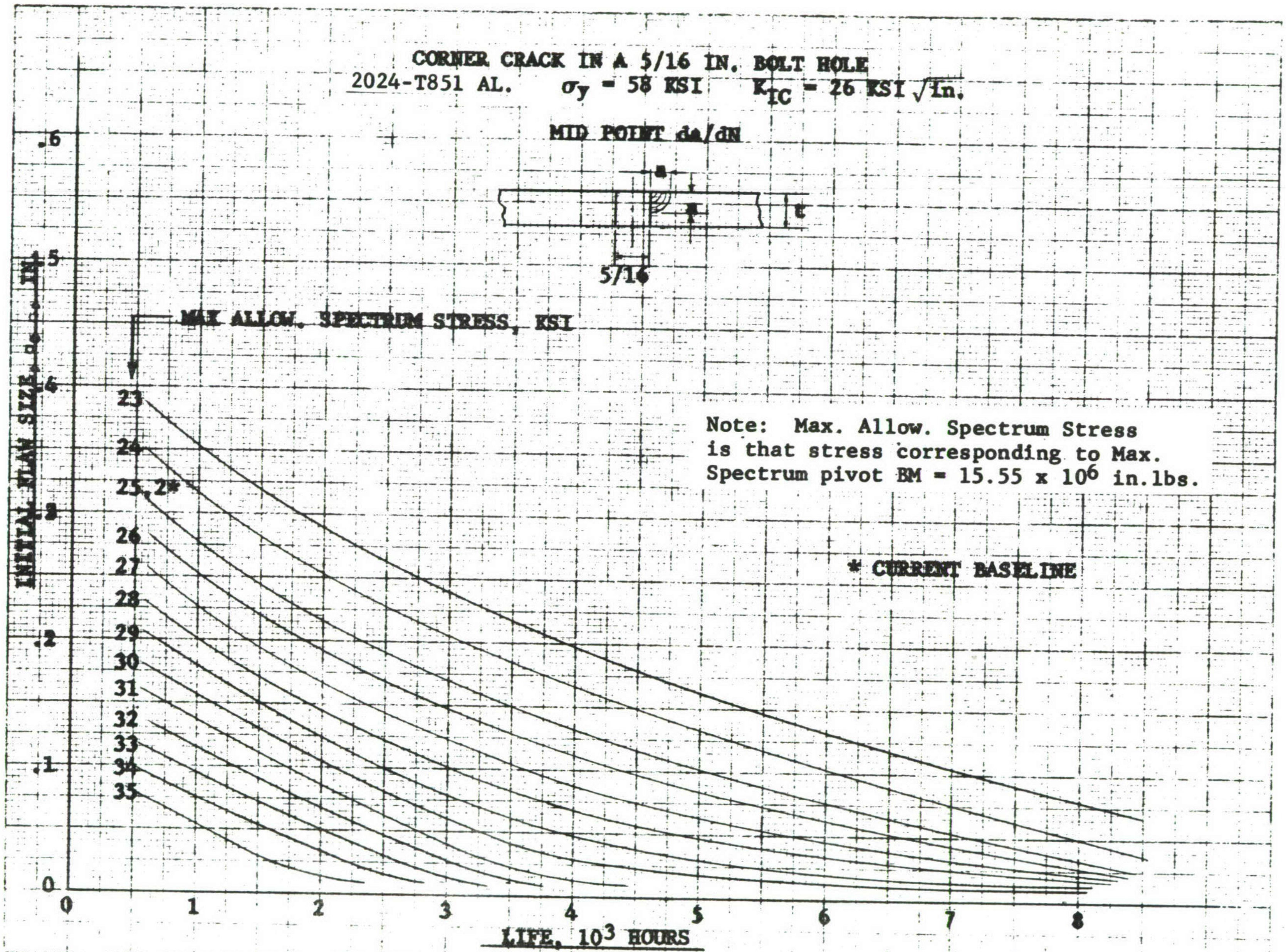


Figure 39 Effect on Life of Constant Allowable Stress

The weight variation of the baseline wing box is presented in Figure 40 as a function of lower wing skin maximum allowable spectrum design stress level. The current maximum baseline lower skin stress level corresponding to maximum fatigue spectrum pivot bending moment is 25.2 ksi. The current weight of one baseline wing is therefore about 1550 pounds. The delta weight penalty is simply the difference between 1550 pounds and the new weight determined from Figure 40 for stresses dictated by the damage tolerance requirements.

A discussion of the redesigned baseline and the corresponding delta weight for use in the basic ADP Wing Contract is given in Section IX.6.

As previously stated, the construction of Figure 40 is based on techniques developed for use with the "Analytical Assembly" phase of the basic contract. A brief discussion is given below to illustrate this approach.

Baseline "Analytical Assemblies" (AA's) of constant cross-section, and forty eight inches in span were carefully sized to provide accurate weight calculations along the span at baseline center spar stations (C.S.S.) 140 and 340. Sizing of these AA's reflected the required lower surface stress level changes due to preliminary design fracture allowables for C.S.S. 140. Fracture allowables for C.S.S. 340 have no impact due to the smaller wing loads at this outboard station.

The basic structural weights for one wing were calculated using weight data directly from the AA's. This procedure calculates 1197 pounds based on the current baseline design, and 1315 pounds when the C.S.S. 140 fracture penalty is included. These weights are a direct function of the lower surface design stresses at C.S.S. 340 and 140. This same procedure (assessing a fracture penalty at C.S.S.140) enabled the calculation of additional points reflecting weight variations with respect to lower surface design stress levels. These calculated weights and stresses were then plotted as shown in Figure 40.

The accuracy of this procedure generally compares well to the more tedious method of calculating weights for unit cross-sections at various points along the span, plotting the data, and integrating the curve to obtain total wing weights. This was done as a check for the design stresses used to size the

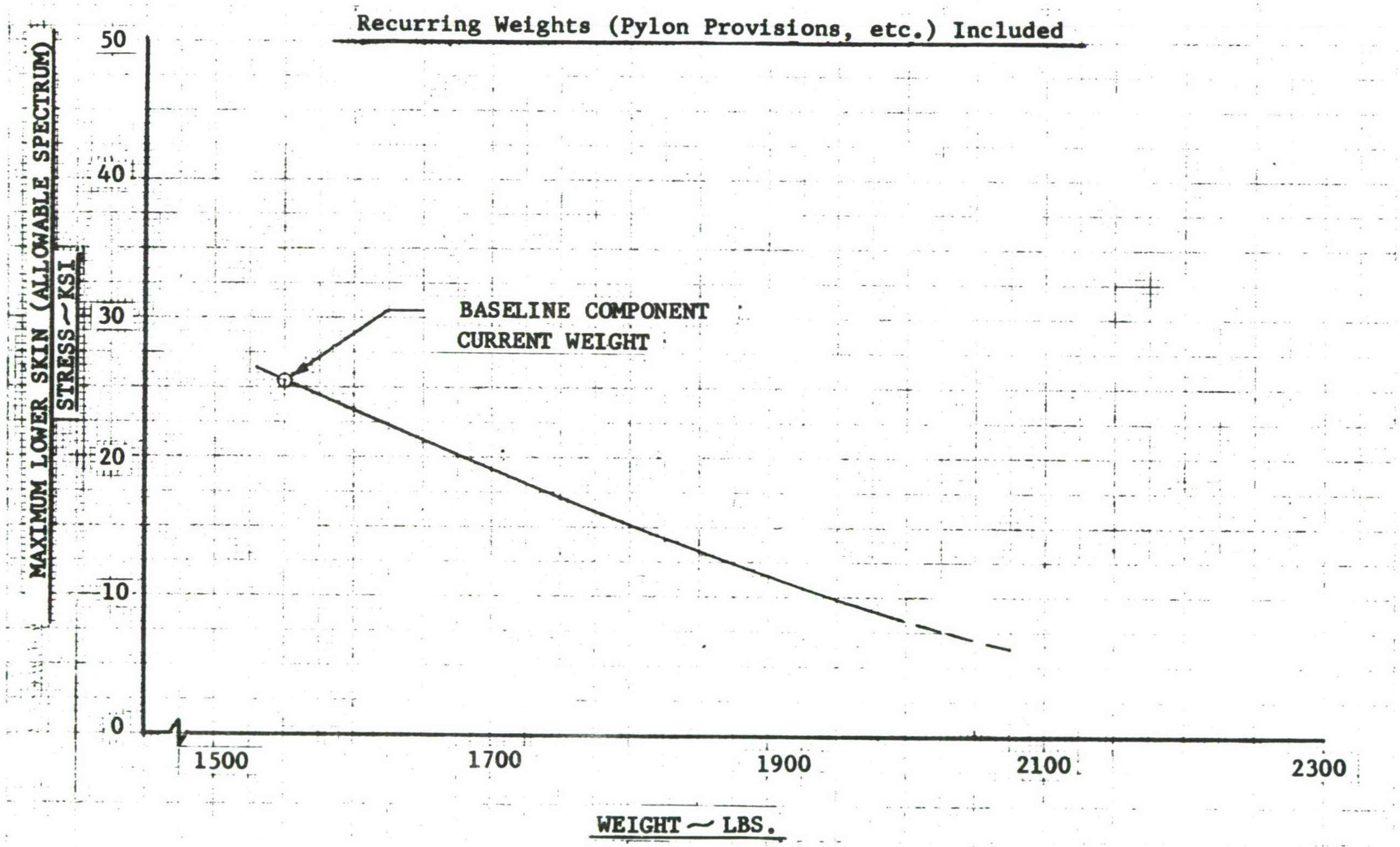


Figure 40 Effect of Lower Skin Design Stress on Weight

previously described AA's. The results are shown in Figure 41. Total wing weights of 1193 pounds versus 1197 pounds for no fracture penalty, and 1315 pounds versus 1303 pounds for the C.S.S. 140 stress penalty were obtained.

While the results of the above comparison agree very well, the basic assumption that the fracture penalty established for C.S.S. 140 will apply to the entire inboard span is conservative because stresses are highest at C.S.S. 140. Ignoring this conservation is expedient and should not affect the relative comparisons of delta weight resulting from allowable stress levels dictated by variation in the parameters considered in these studies. However, the final weight for the redesigned baseline (Section IX.6) will be calculated in a more exact manner.

It has also been assumed that recurring weights, such as pylon provisions, etc., will be constant. Whether or not this is exactly correct, it enables a consistent appraisal of the variations of weight of the more critical lower skin. The recurring weights for the baseline have been established as a total of 355 pounds and are included in the curve of Figure 40.

Constant Recurring Weights Not Shown

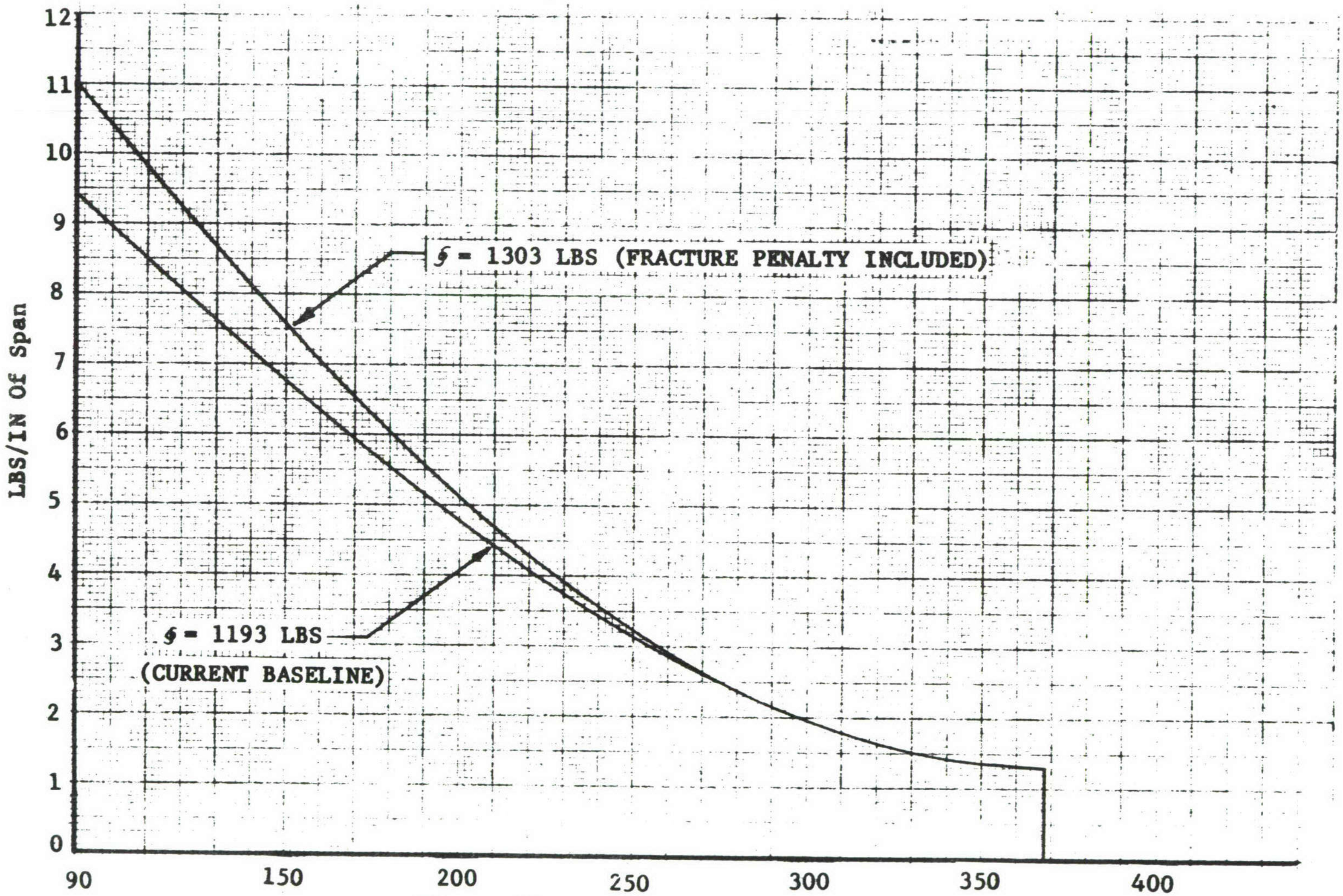


Figure 41 Baseline Wing Structure Weight

IX.3.2 Effects of Varying K_{Ic} and da/dN

IX.3.2.1 Impact of K_{Ic} Variation

The impact of 2024-T851 K_{Ic} variation on allowable stresses may be determined from the curves in Figures 16 through 30. A summary of design allowable stresses for 2000 and 8000 flight hours is given in Table VIII for three typical flaw situations. This summary is based on assuming initial flaw sizes as currently specified by the proposed criteria requirements.

An indication of K_{Ic} variation on life is given in Table IX for arbitrarily selected constant stress levels. The data in the table was taken from the curves shown in Figures 31 through 39. Similar data can be obtained from these curves at other stress levels.

The impact of lower, middle, and upper K_{Ic} variation on critical flaw sizes is given in Figures 42 through 46 as plots of stress versus critical flaw size. The critical flaw sizes used in developing design allowable curves for this study were determined from these curves.

Table V in paragraph IX.3.1 presents the data considered in selecting the lower (20 ksi $\sqrt{in}$), mid-point (23 ksi$\sqrt{in}$), and upper bound (26 ksi $\sqrt{in}$) K_{Ic} values.

IX.3.2.2 Impact of da/dN Variation

The impact of varying da/dN to include the upper and lower bounds of the data has been assessed for a typical part through flaw in the lower skins at C.S.S. 140 (t = .611"), and for both a through the thickness and corner type bolt hole flaw in 5/16" diameter fastener holes. The assessment is presented as allowable curves for life intervals of 2000, 4000, and 8000 flight hours in Figures 47 through 52. Mid-point fracture toughness data (K_{Ic} = 23 ksi $\sqrt{in}$) was used in developing these curves. The da/dN data used in this study is that previously given in Figure 9. A typical summary of design allowable stresses resulting from da/dN variation is given in Table X for life intervals of 2000 and 8000 hours. This summary is based on assuming initial flaw sizes typical of those currently specified in the criteria. Similar data can be quickly obtained for other initial flaw sizes using the curves in Figures 47 through 52.

Table VIII

IMPACT OF K_{IC} VARIATION
ON DESIGN ALLOWABLE STRESS LEVEL

2024-T851 AL F-111 Baseline Severe Usage

Mid-Point da/dn Data

FLAW DESCRIPTION	ASSUMED INITIAL FLAW SIZE	MAX. ALLOWABLE SPECTRUM STRESS LEVELS, KSI					
		K_{IC} = 20 $KSI\sqrt{IN}$		K_{IC} = 23 $KSI\sqrt{IN}$		K_{IC} = 26 $KSI\sqrt{IN}$	
		2000 HR.	8000 HR.	2000 HR.	8000 HR.	2000 HR	8000 HR
SURFACE FLAW-PART THROUGH t = .611 , a/2c = .5	a/Q = .1 a_o = .246	22.4	19.2	24.4	19.7	25.9	19.9
BOLT HOLE FLAW-THROUGH THE THICKNESS 5/16 DIA	a_o = .05	18.2	16.2	20.3	16.9	21.5	17.9
BOLT HOLE FLAW-CORNER CRACK 5/16 DIA	a_o = .01	32.5	29.6	34.4	29.8	35.3	29.8

Table IX

IMPACT OF K_{IC} VARIATION
ON LIFE AT CONSTANT STRESS LEVEL

2024-T851 Al. F-111 Baseline Severe Usage
Mid-Point da/dN Data

FLAW DESCRIPTION	ASSUMED INITIAL FLAW SIZE	LIFE INTERVAL IN FLIGHT HOURS		
		K_{IC} = 20 KSI $\sqrt{IN.}$	K_{IC} = 23 KSI $\sqrt{IN.}$	K_{IC} = 26 KSI $\sqrt{IN.}$
SURFACE FLAW--PART THROUGH t = .611, a/2c = .5 σ = 25.2 ksi	a/Q = .1 a_o = .246	250	1550	2350
BOLT HOLE FLAW--THROUGH THE THICKNESS 5/16 DIA. σ = 19.0 ksi	a_o = .05	800	3550	5800
BOLT HOLE FLAW--CORNER CRACK 5/16 DIA σ = 32.0 ksi	a_o = .01	2350	3250	3650

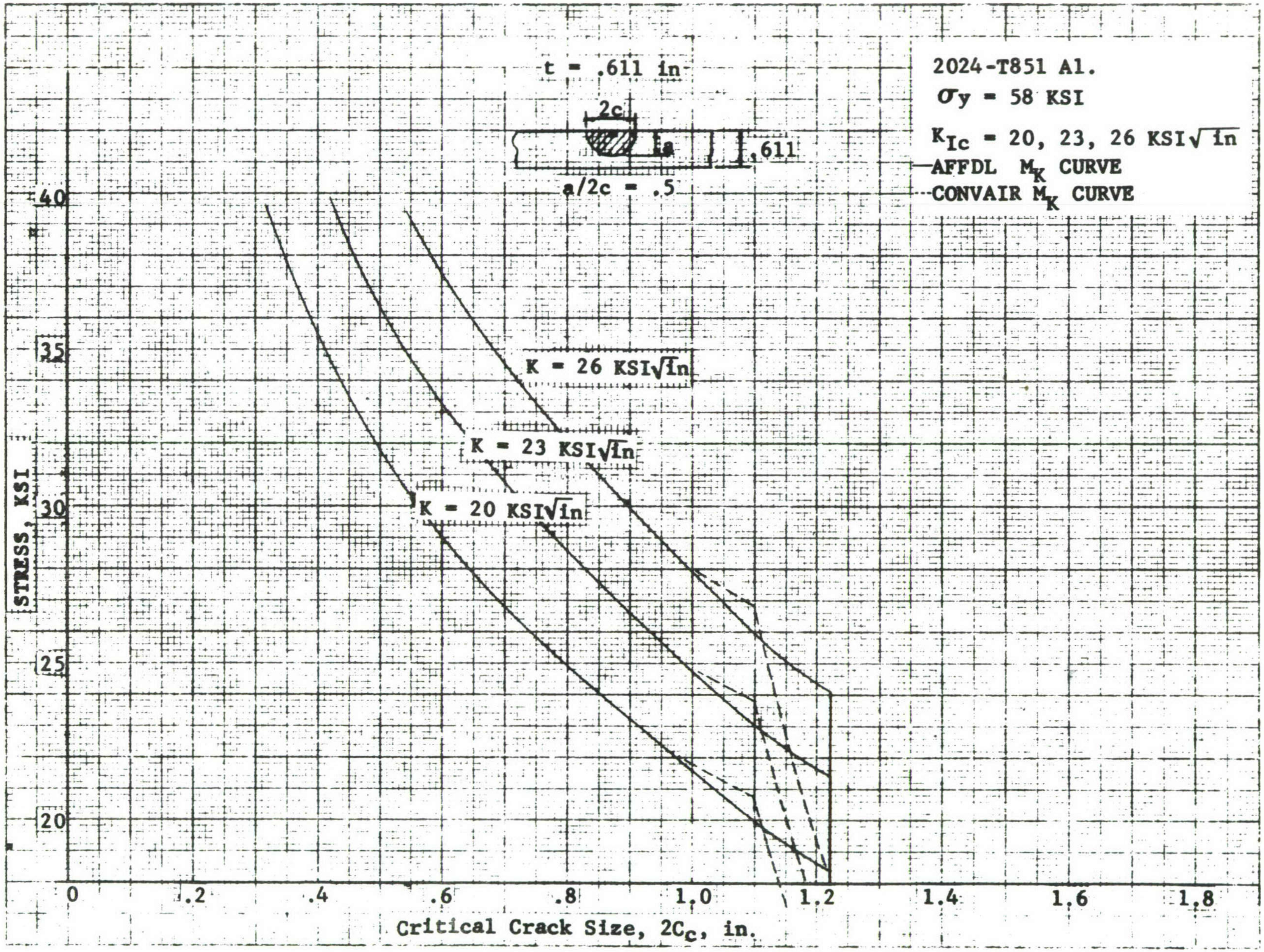


Figure 42 Surface Flaw Critical Crack Sizes ($2C_c$)

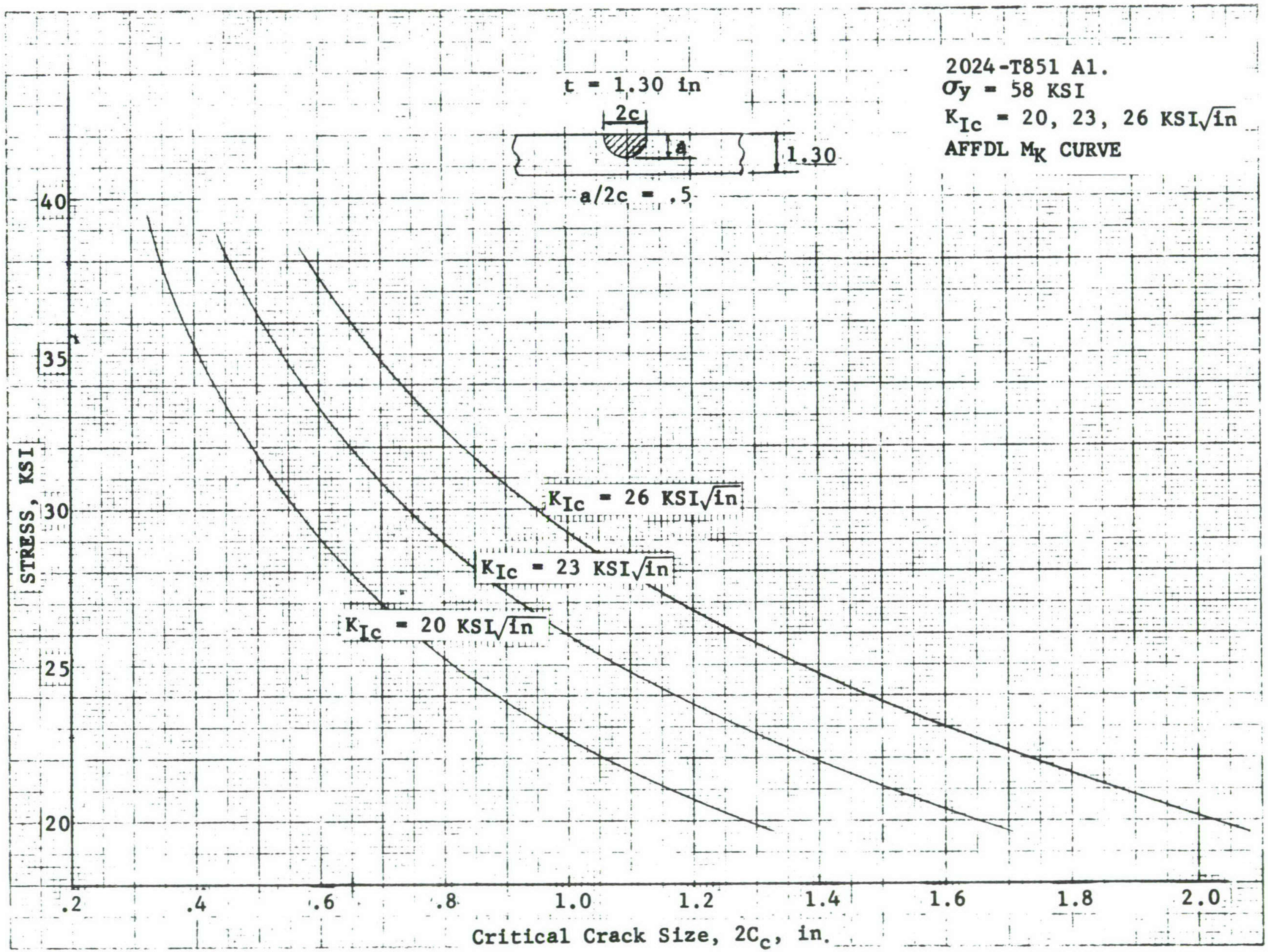


Figure 43 Surface Flaw Critical Crack Sizes ($2C_c$)

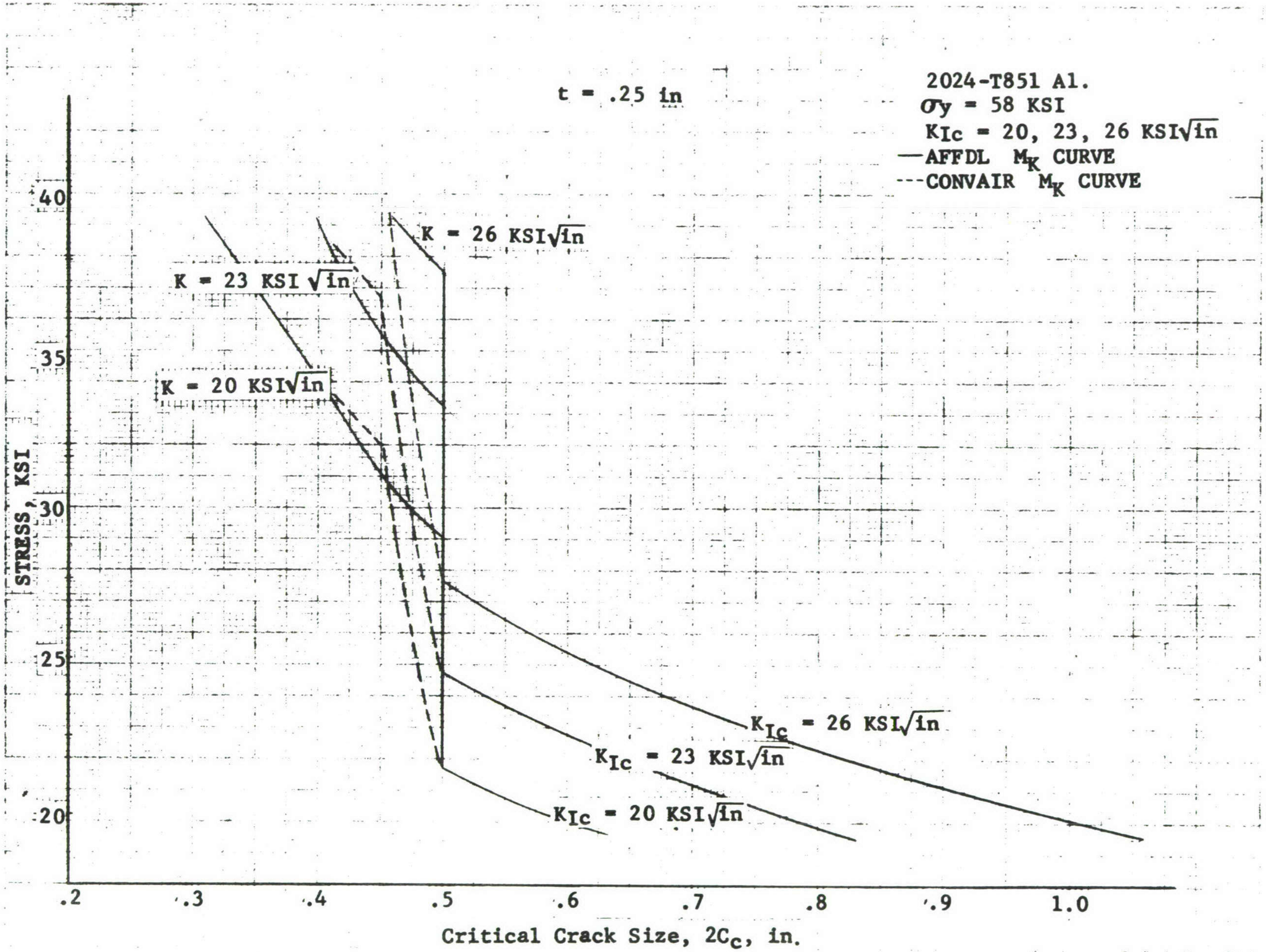


Figure 44 Surface Flaw Critical Crack Sizes ($2C_c$)

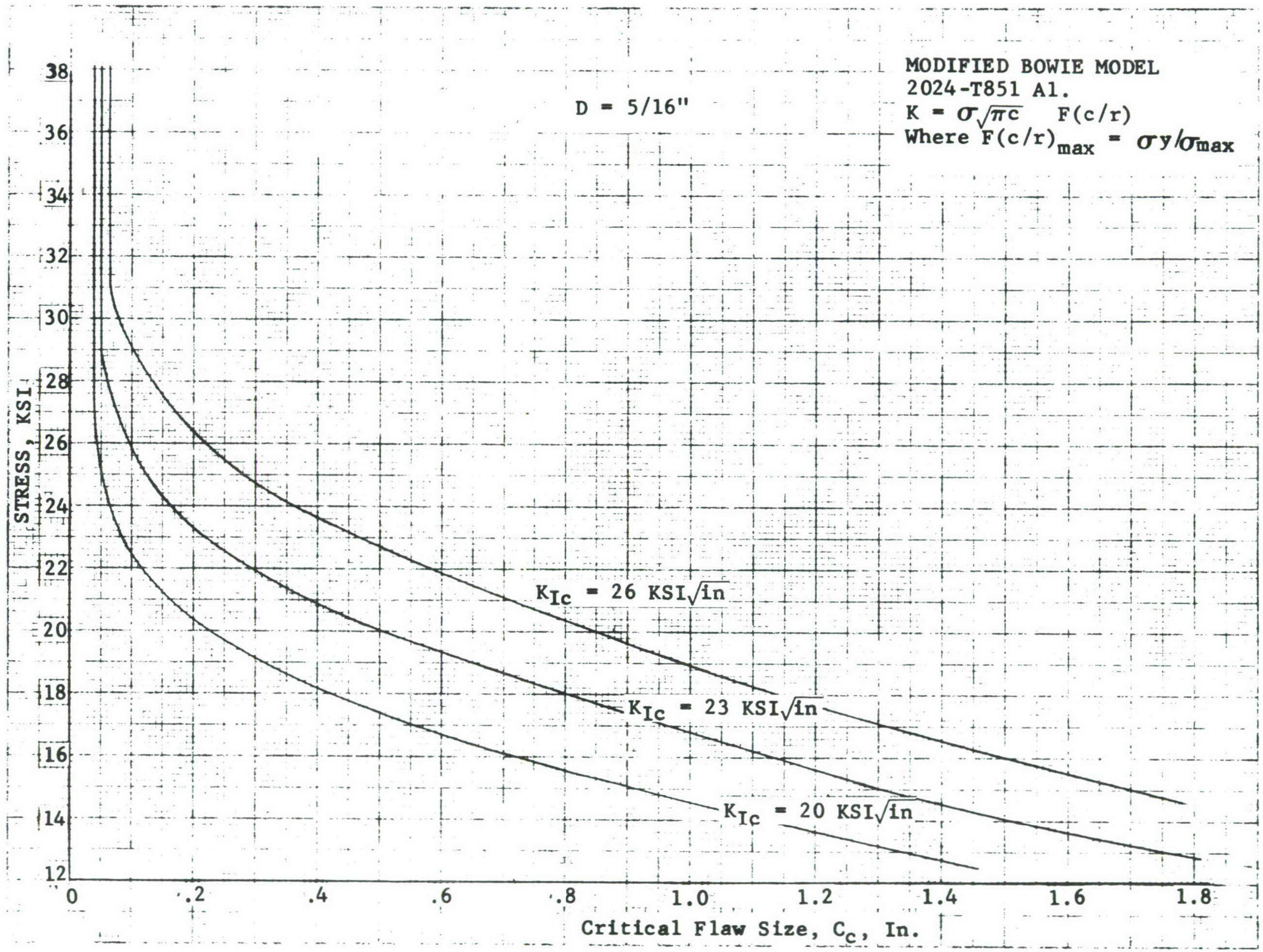


Figure 45 Through the Thickness @ A Hole Critical Flaw Sizes

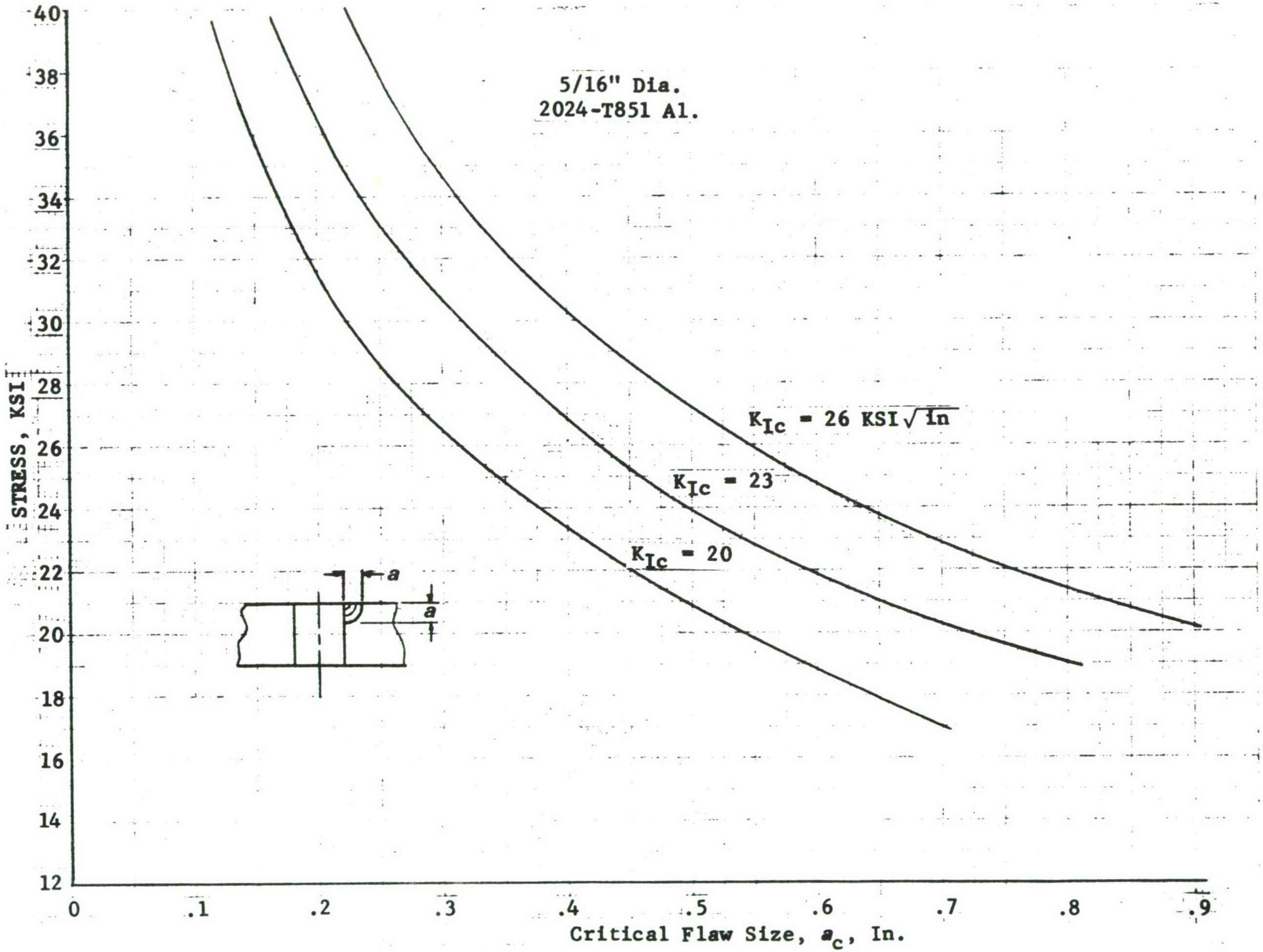


Figure 46 Semi-Circular Corner Crack at A Hole Critical Flaw Sizes

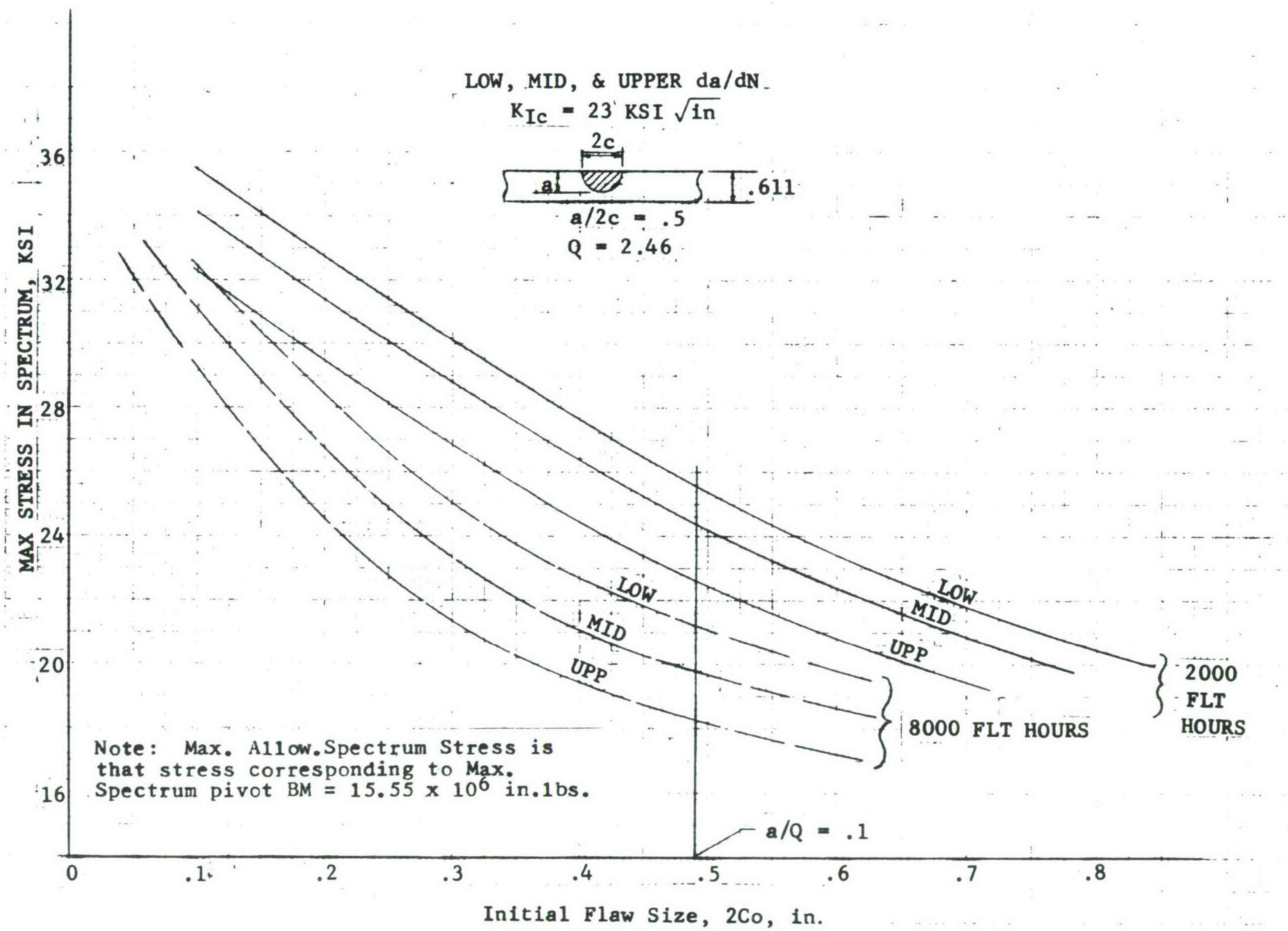


Figure 47 Allowable Curves for 2024-T851 Surface Flaws in .611 Skin

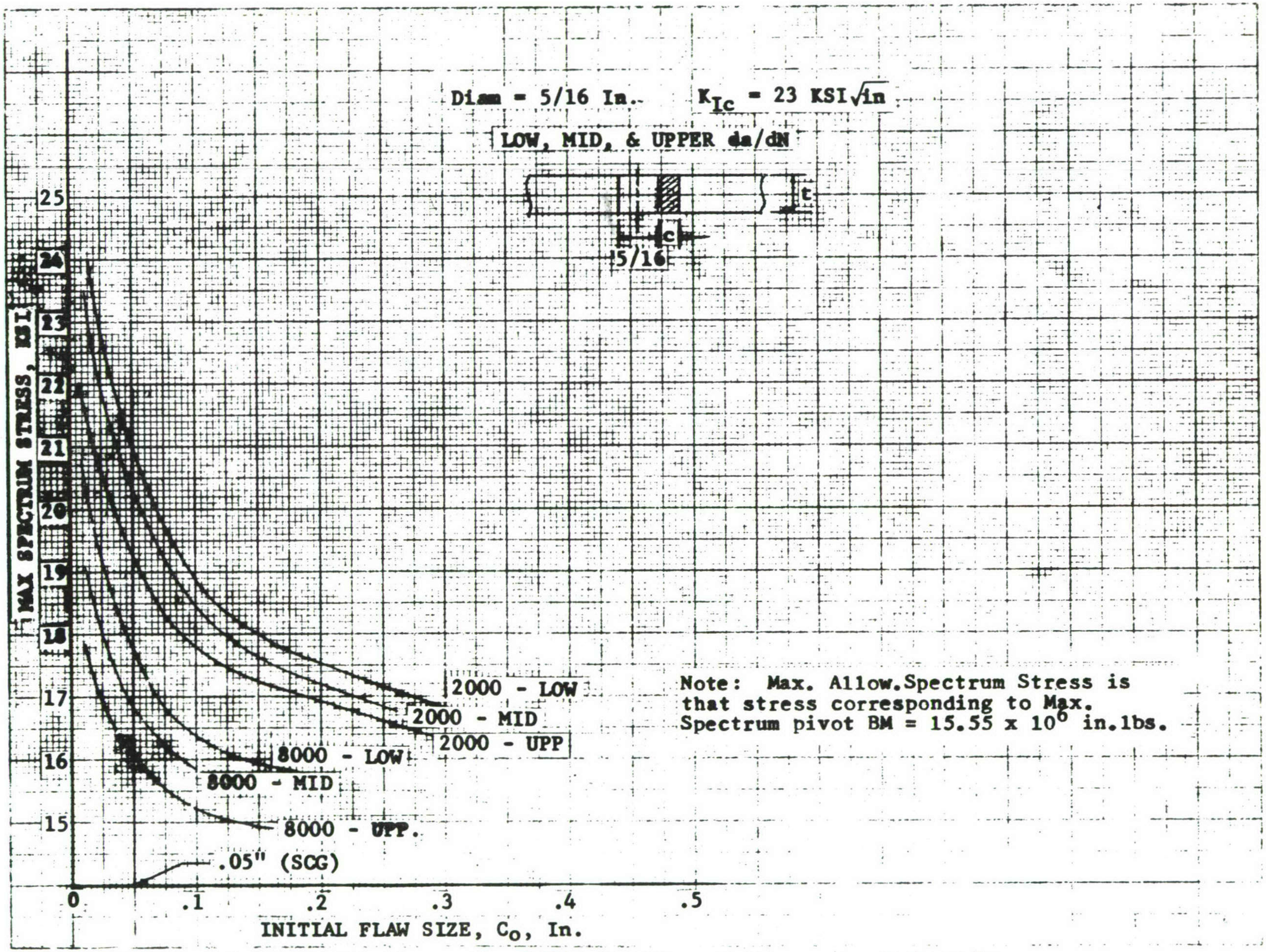


Figure 48 Allowable Curves for 2024-T851 Through Flaws at a Bolt Hole

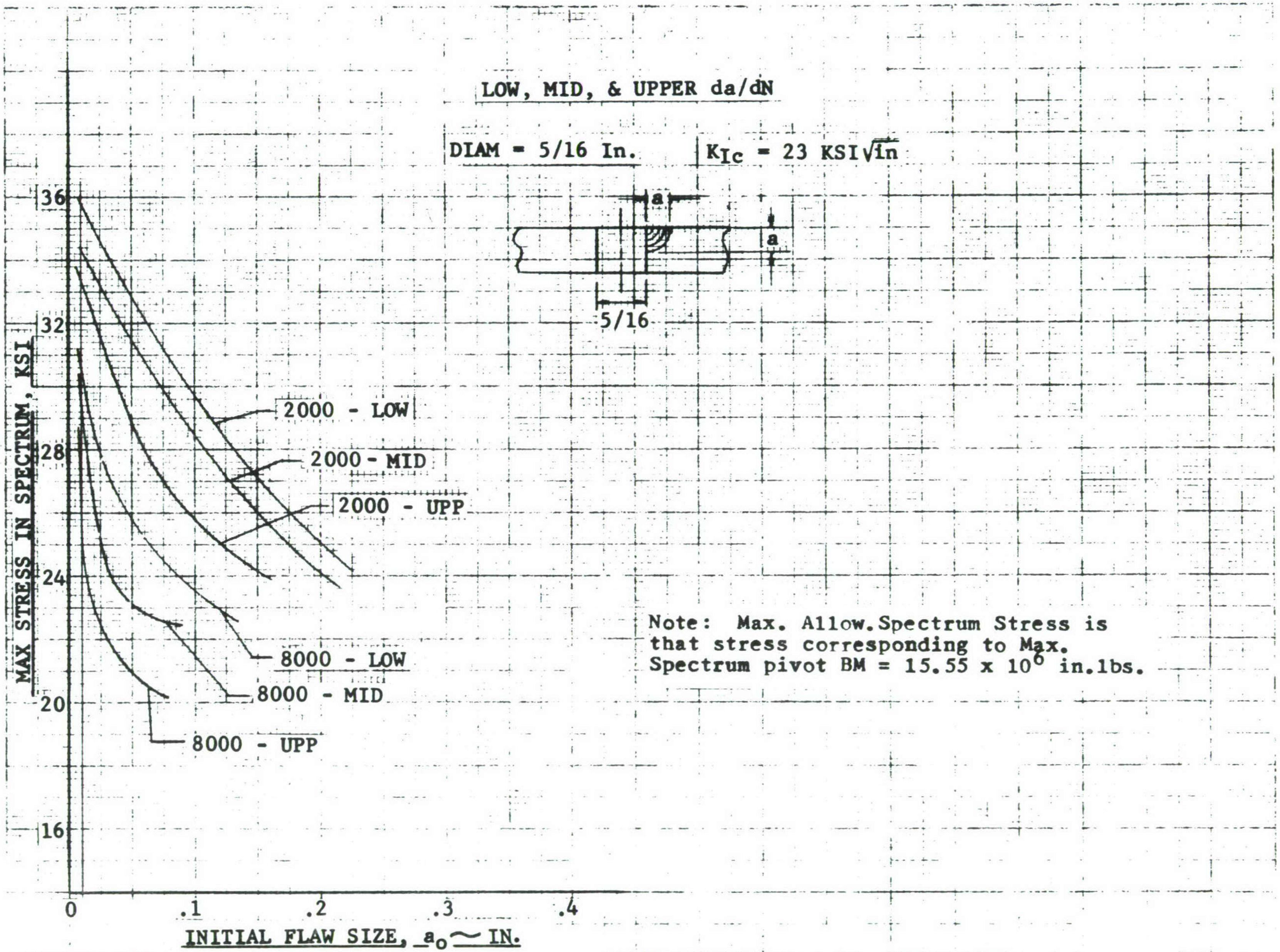


Figure 49 Allowable Curves for 2024-T851 Semi-Circular Corner Crack at a Hole

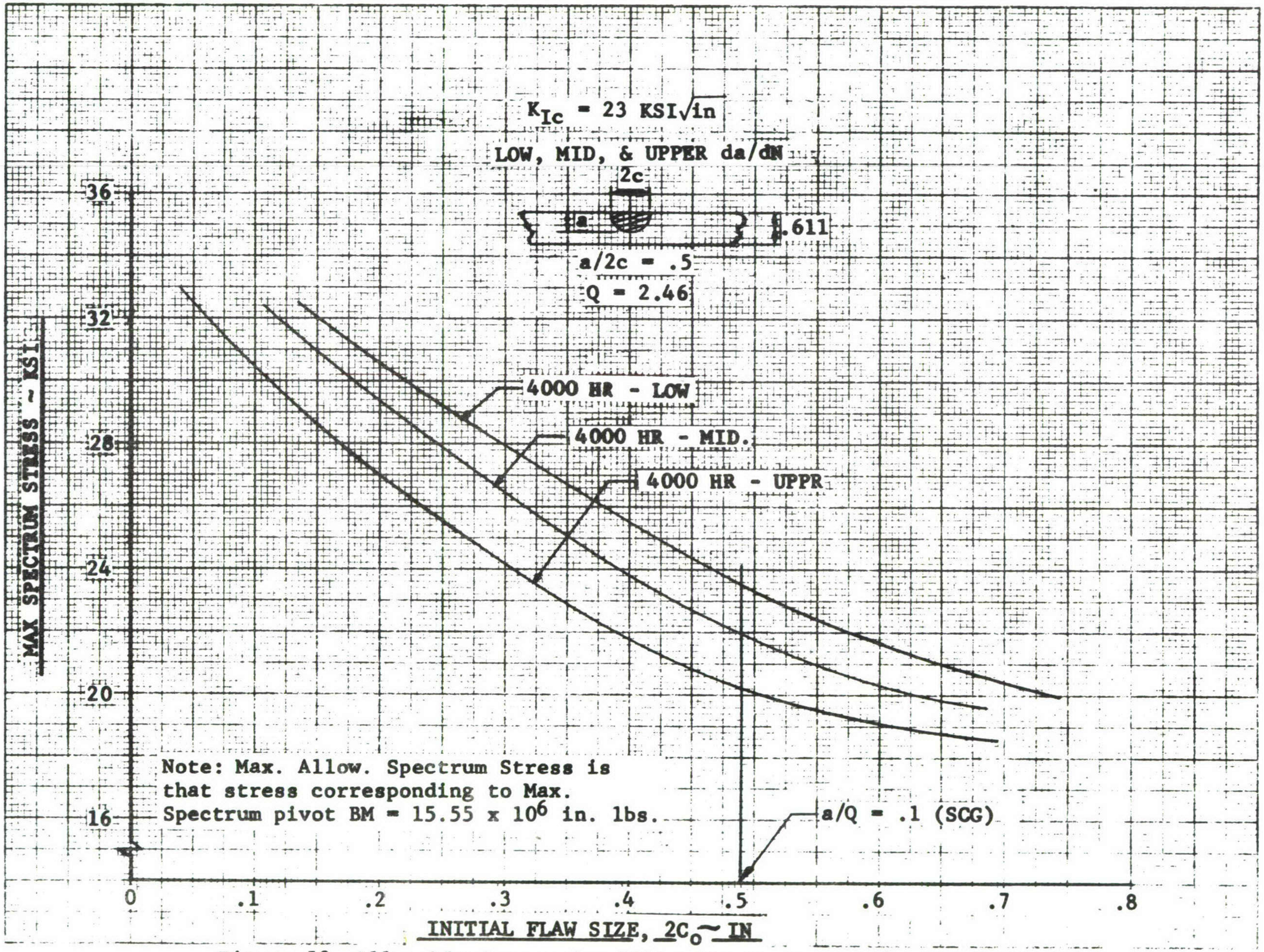


Figure 50 Allowable Curves for 2024-T851 Surface Flaws in 0.611 Skin

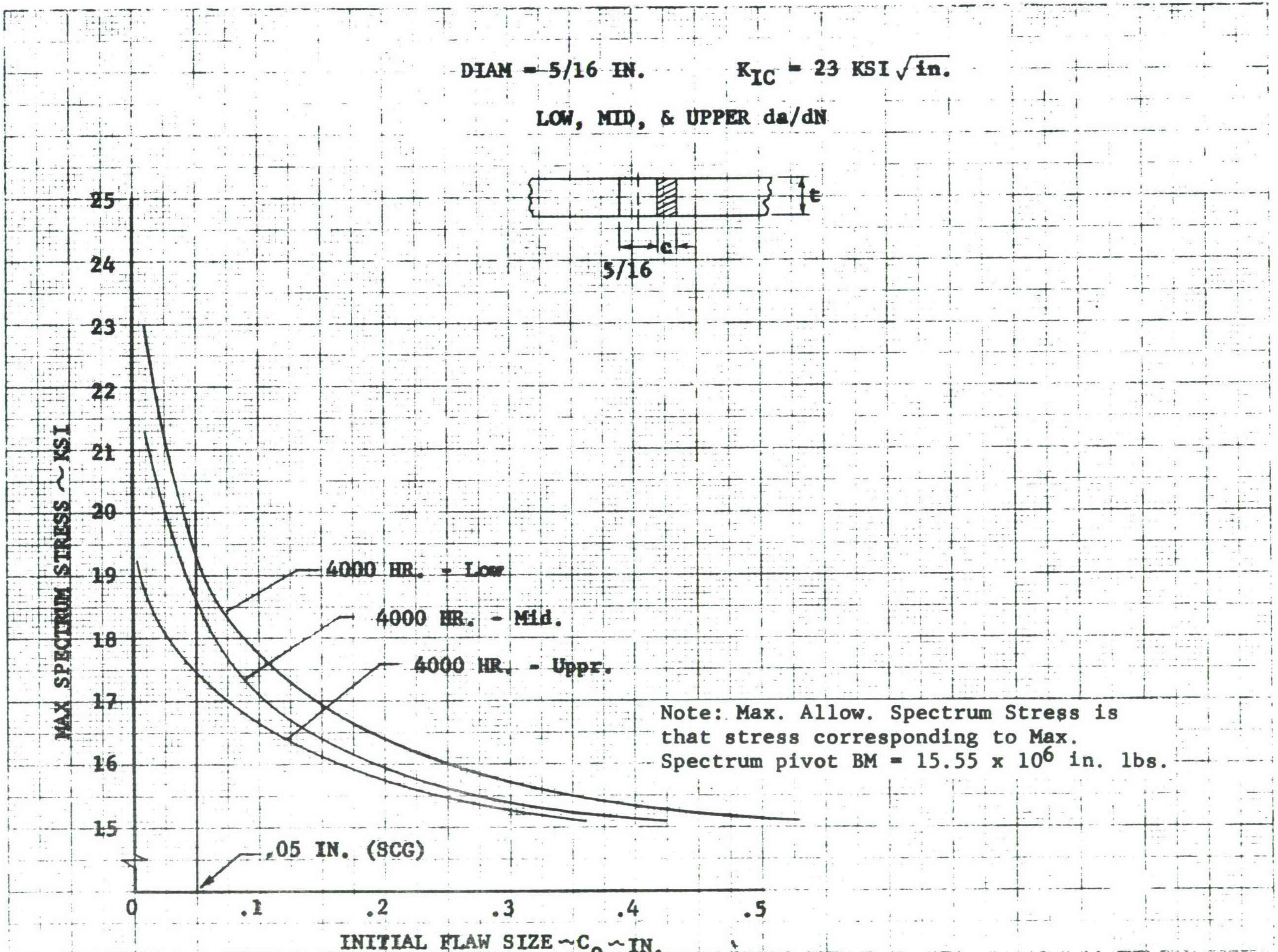


Figure 51 Allowable Curves for 2024-T851 Through Flaws at a Bolt Hole

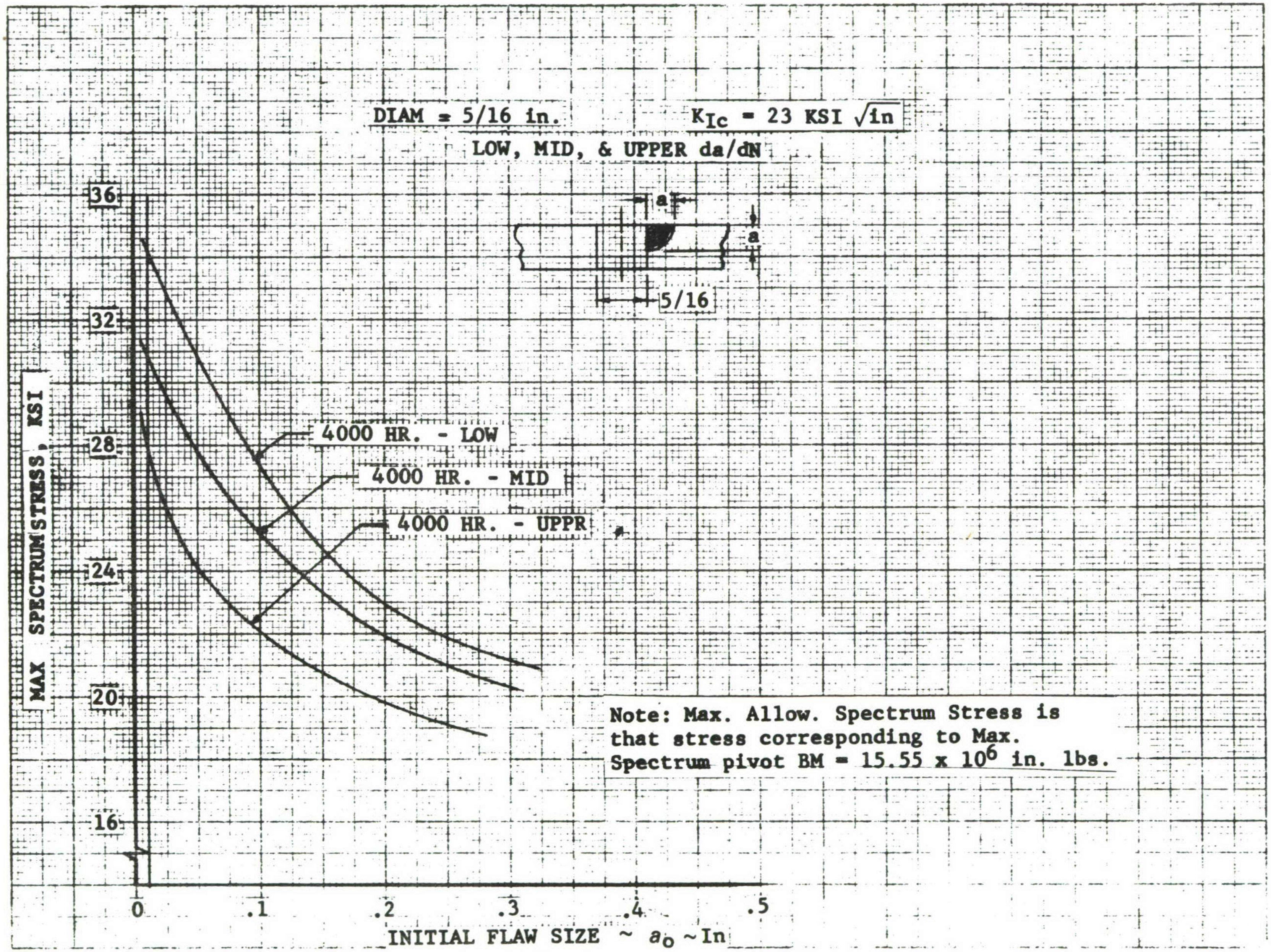


Figure 52 Allowable Curves for 2024-T851 Semi-Circular Corner Crack at a Hole

Table X

IMPACT OF da/dN VARIATION
ON DESIGN ALLOWABLE STRESS LEVEL

2024-T851 AL F-111 Baseline Severe Usage

FLAW DESCRIPTION	ASSUMED INITIAL FLAW SIZE	MAX. ALLOWABLE SPECTRUM STRESS LEVELS, KSI					
		LOW da/dN DATA		MID da/dN DATA		UPP da/dN DATA	
		$K_{IC} = 23$ KSI $\sqrt{IN}$		$K_{IC} = 23$ KSI $\sqrt{IN}$		$K_{IC} = 23$ KSI $\sqrt{IN}$	
		2000 HR.	8000 HR.	2000 HR.	8000 HR.	2000 HR	8000 HR
SURFACE FLAW-PART THROUGH $t = .611$, $a/2c = .5$	$a/Q = .1$ $a_o = .246$	25.6	21.2	24.4	19.8	22.6	18.3
BOLT HOLE FLAW-THROUGH THE THICKNESS 5/16 DIA	$a_o = .05$	21.0	17.8	20.3	16.9	19.3	16.1
BOLT HOLE FLAW-CORNER CRACK 5/16 DIA	$a_o = .01$	35.8	30.6	34.4	29.8	33.4	25.2

To illustrate the effect of da/dN variation on life for a constant stress level the data in Figures 47 and 48 for the surface flaw (t = .611") was cross-plotted by entering the curves at several selected stress levels and determining the initial flaw sizes for 2000, 4000, and 8000 flight hours. The resulting "Flaw size versus Life" curves are shown in Figures 53 through 55 for upper, lower and mid-point da/dN data. These curves were then used to determine the life interval for arbitrarily selected typical constant stress levels as summarized in Table XI. The identical procedure would allow assessment of other constant stress levels on life for variation in da/dN data. The same procedure can be applied to the bolt hole flaw types.

There has been some interest expressed in the non-linear effects on life when the "C" coefficient in the Forman equation is increased or decreased by some amount. The data presented in Table XI provides such information since the upper and lower bounds to the da/dN data used in this study were determined by factoring the "C" coefficient. See paragraph IX.3.1.1 for a previous discussion of the crack growth data. The upper bound da/dN curve represents a 1.45 factor on "C", and the lower bound curve represents a factor of 0.766 on "C". The results in Table XI indicate that a reduction in growth rate was more significant than an increase in the growth rate.

IX.3.3 Effects of Varying Usage Spectrum

The effects on life and crack growth allowable stress for severe (F-111 Phase I and II Training) usage and mild (F-111 recorder data) usage were determined.

The approach to assess the impact of mild usage was development of design stress allowable curves similar to those developed in paragraph IX.3.1 for severe usage. Mid-point K_{Ic} and da/dN data were used, and other analysis assumptions are identical to those described previously.

Results indicating the impact on allowable stress levels are shown in Table XII and in Figures 56 through 58 for the following flaw types:

(1) Part through surface flaw in the lower skin at C.S.S. 140 (t = .611").

(2) Through the thickness flaw at a 5/16 diameter bolt hole.

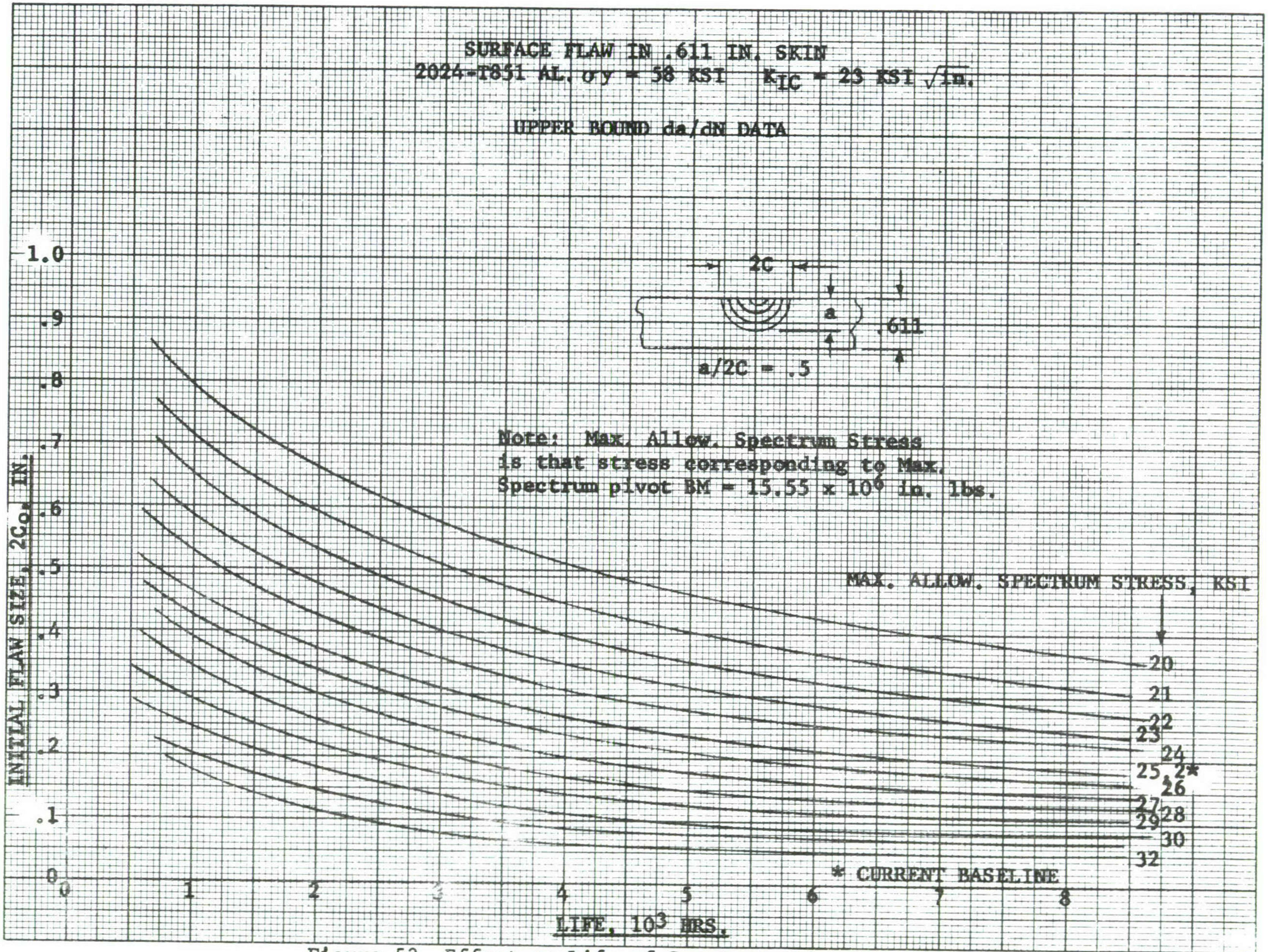


Figure 53 Effect on Life of Constant Allowable Stress

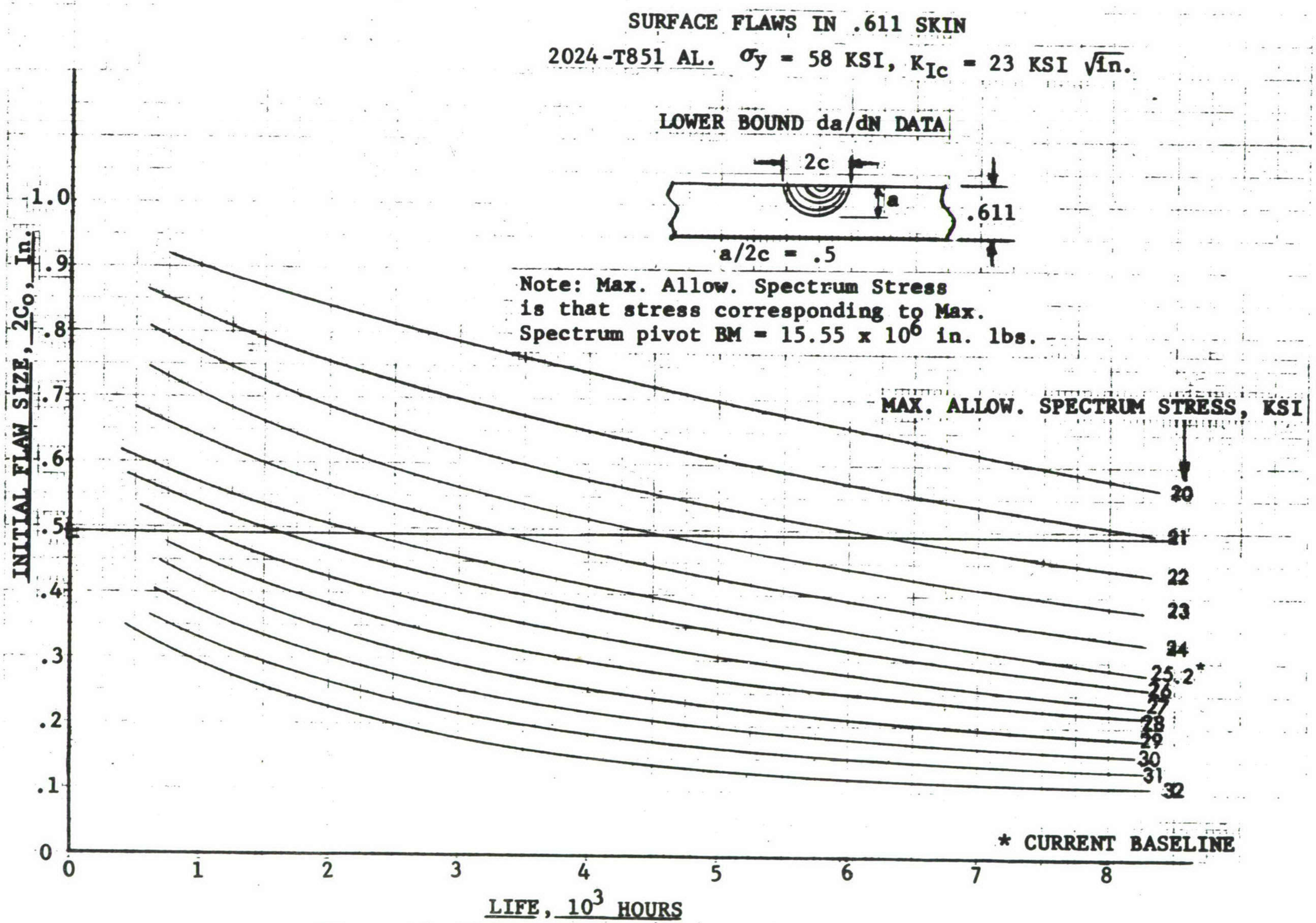


Figure 54 Effect on Life of Constant Allowable Stress

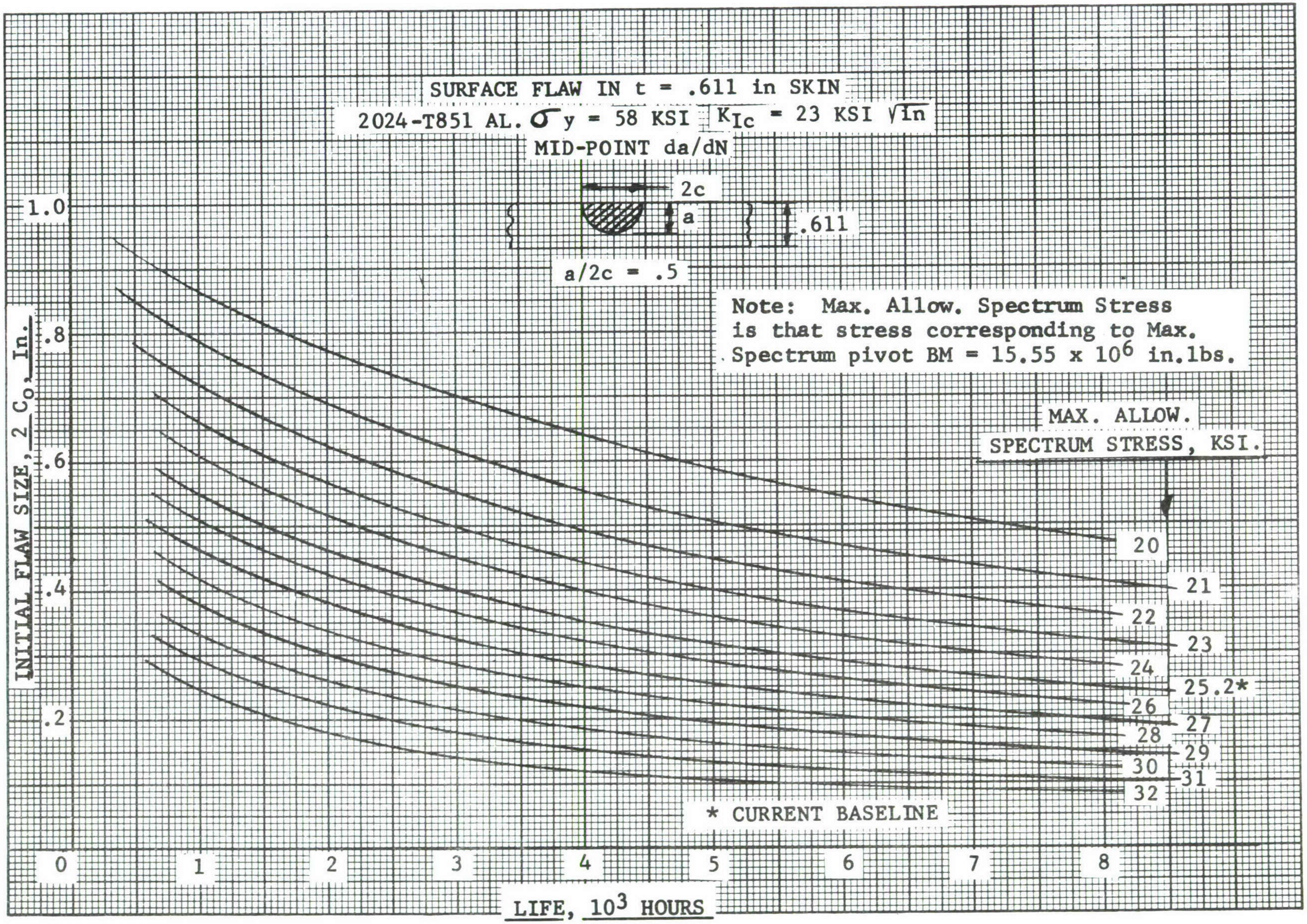


Figure 55 Effect on Life of Constant Allowable Stress

Table XI

IMPACT OF da/dN VARIATION
ON LIFE AT CONSTANT STRESS LEVEL

2024-T851 Al. F-111 Baseline Severe Usage
Mid-Point K_{IC} Data (23 ksi $\sqrt{in.}$)

FLAW DESCRIPTION (STRESS LEVEL)	ASSUMED INITIAL FLAW SIZE	LIFE INTERVAL IN FLIGHT HOURS		
		LOW da/dN DATA	MID da/dN DATA	UPP da/dN DATA
SURFACE FLAW--PART THROUGH t = .611, a/2c = .5 (σ = 22 ksi)	a/Q = .1 a_o = .246	6250	3850	2350
SURFACE FLAW--PART THROUGH t = .611, a/2c = .5 (σ = 25.2 ksi)*	a/Q = .1 a_o = .246	2200	1500	700
SURFACE FLAW--PART THROUGH t = .611, a/2c = .5 (σ = 28 ksi)	a/Q = .1 a_o = .246	600	350	0

* CURRENT BASELINE STRESS LEVEL

Table XII

IMPACT ON DESIGN ALLOWABLE STRESS DUE TO SEVERE AND MILD USAGE

MID-PT. K_{IC} = 23 KSI IN
MID-PT. da/dN DATA

FLAW DESCRIPTION	ASSUMED INITIAL FLAW SIZE	MAX. ALLOWABLE SPECTRUM STRESS LEVELS, KSI					
		SEVERE USAGE*			MILD USAGE		
		2000 HR.	4000 HR.	8000 HR.	2000 HR.	4000 HR.	8000 HR.
SURFACE FLAW-PART THROUGH t = .611 , a/2c = .5	a/Q = .1 a_o = .246	24.4	21.9	19.7	27.6	26.5	24.9
BOLT HOLE FLAW-THROUGH THE THICKNESS 5/16 DIA	a_o = .05	20.3	18.6	16.9	22.9	22.1	20.9
BOLT HOLE FLAW-CORNER CRACK 5/16 DIA	a_o = .01	34.4	30.7	29.8	46.3	43.6	39.2

* Reference Design Allowable Curves in Figures 16 through 30 for Severe Usage.

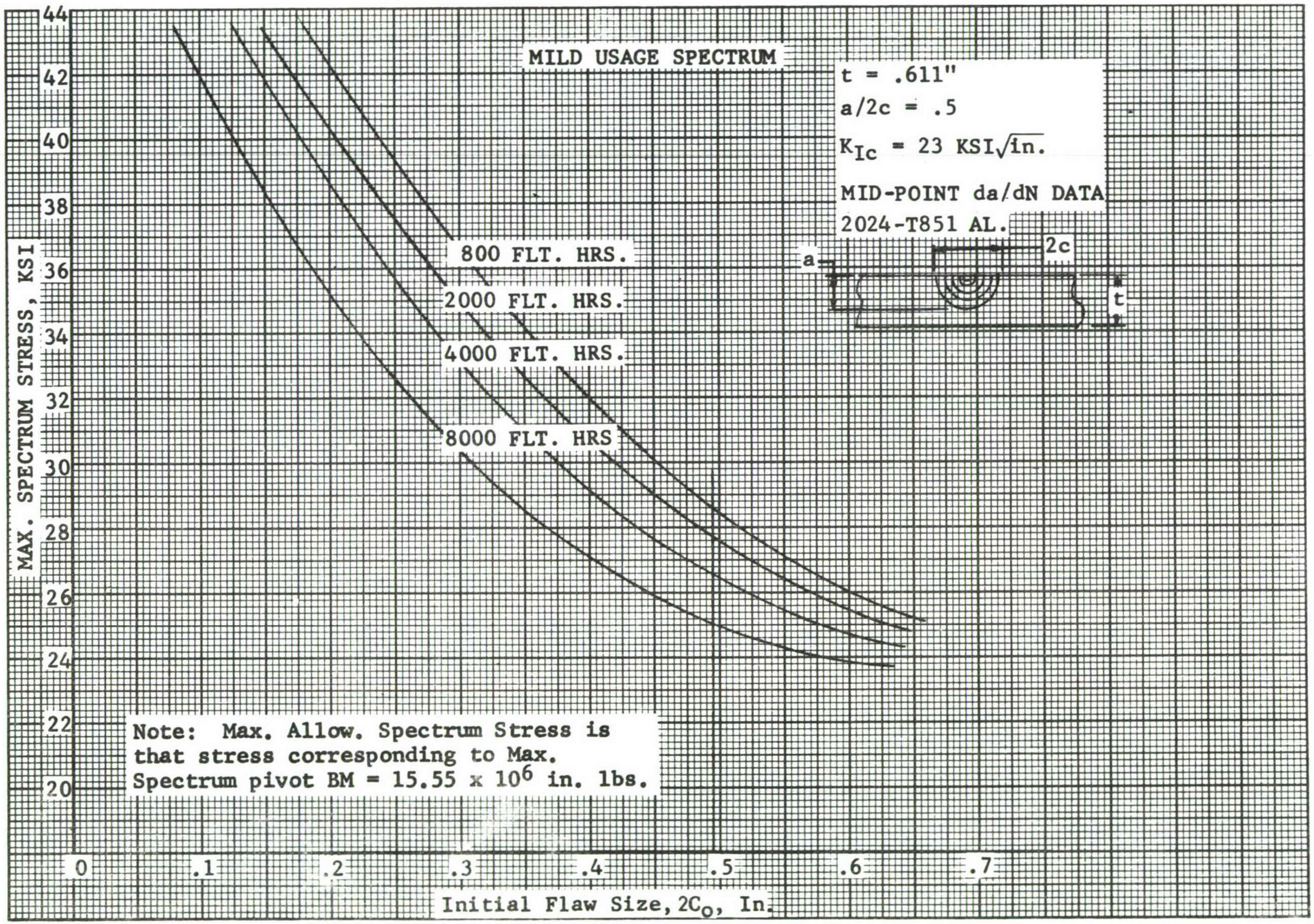


Figure 56 Part Through Surface Flaw Design Allowable Curves

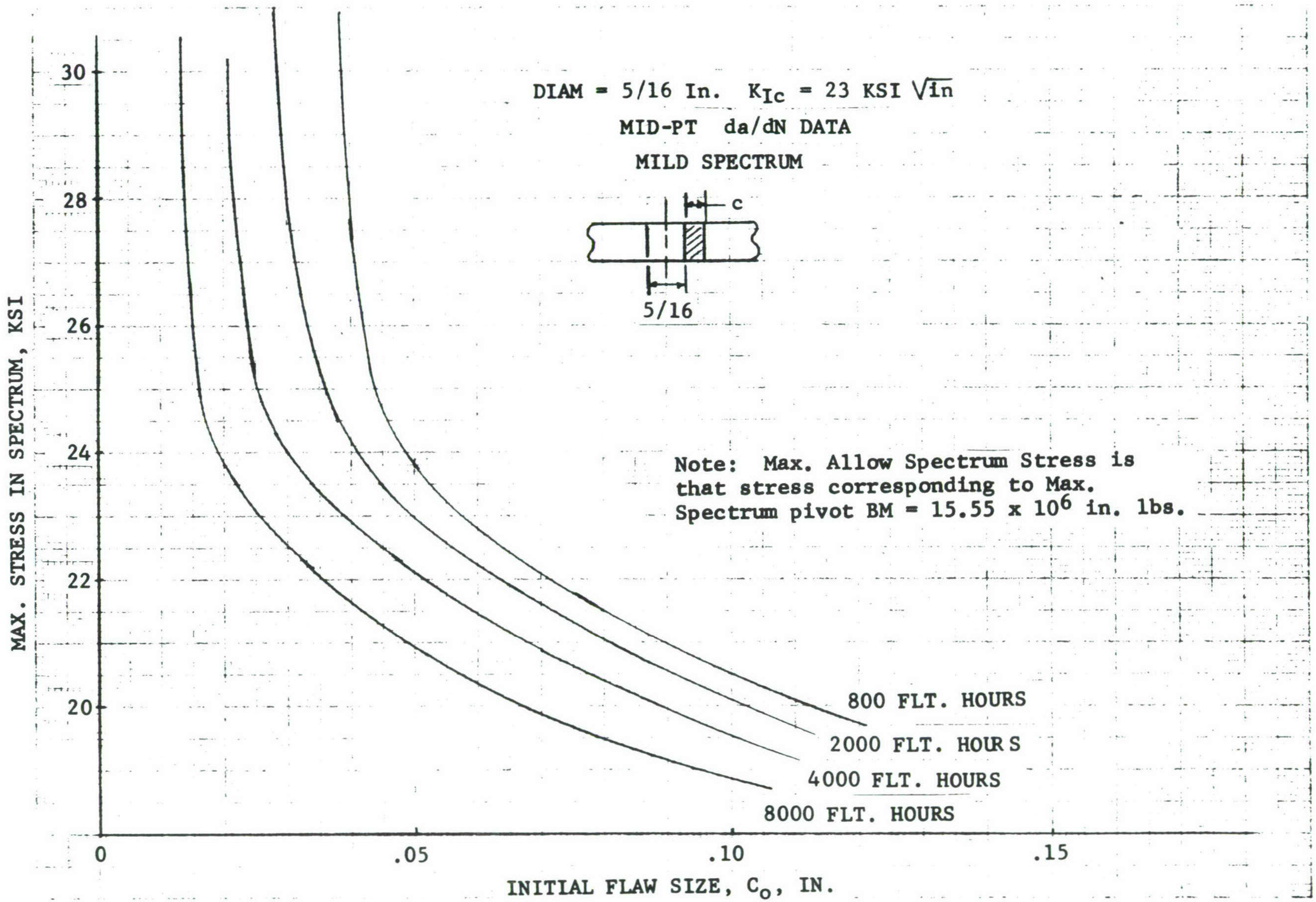


Figure 57 Allowable Curves for 2024-T851 Through the Thickness at a Bolt Hole

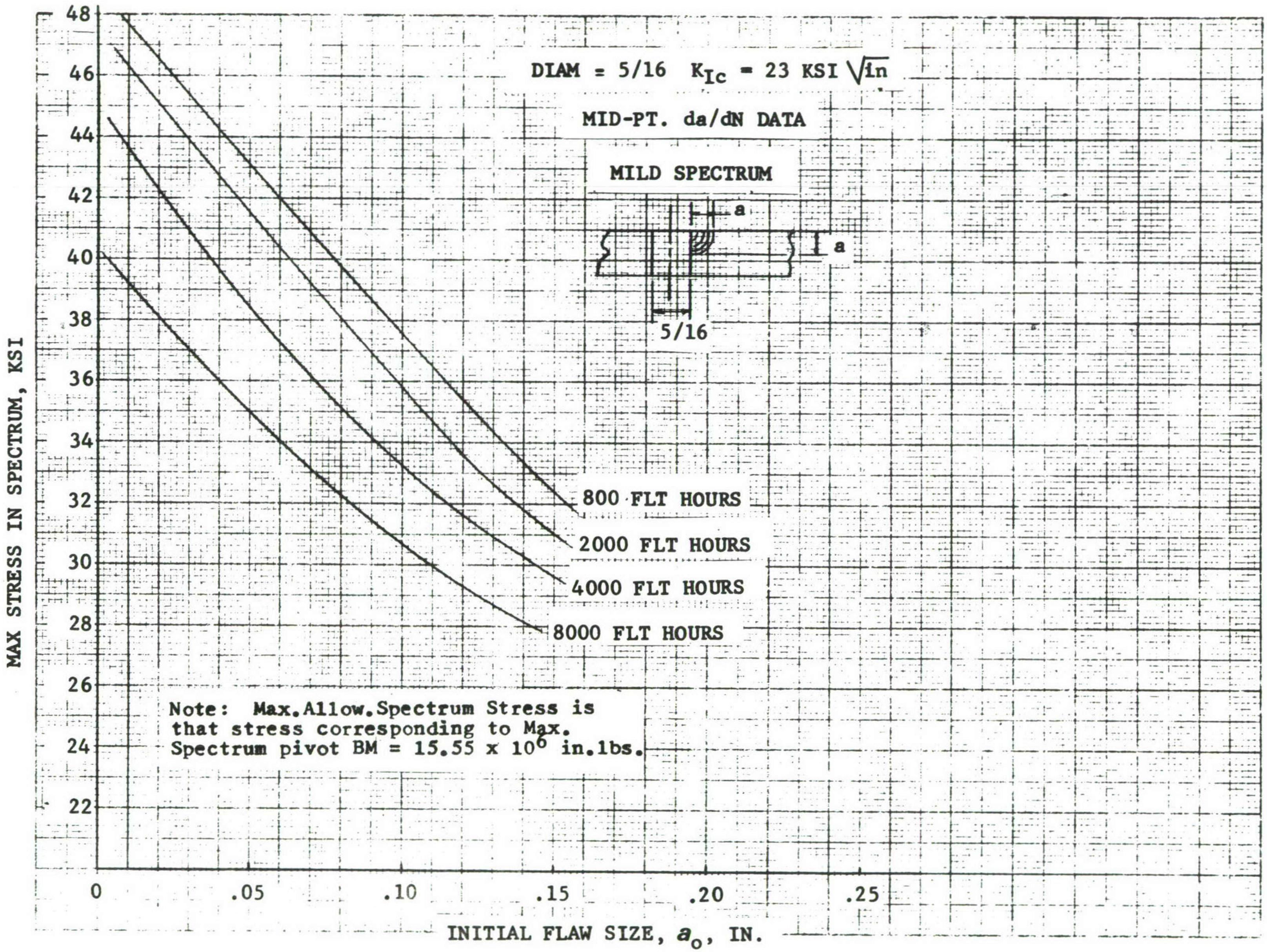


Figure 58 Allowable Curves for 2024-T851 Semi-Circular Corner Crack at a Hole

(3) Semi-circular corner crack at a 5/16 diameter bolt hole.

The variation in maximum allowable spectrum stress level for life intervals of 800, 2000, 4000, and 8000 flight hours may be determined from the Figures for various initial flaw size assumptions other than those used to develop Table XII.

The variation of life for three arbitrarily selected constant stress levels is summarized in Table XIII based on severe and mild usage. The part-through flaw case is shown for illustration. An approximate five-to-one increase in life results from the mild usage spectrum analyses.

The mild usage (F-111 recorder) and severe usage (Phase I and II Training) spectra are shown on the pivot bending moment exceedance plots in Figure 59. The following mission segment mix was established for the mild usage spectrum using comparisons between n_z exceedance plots corresponding to the primary design usage and 500 hours n_z exceedance data obtained from in-flight recorders on TAC F-111 aircraft:

MISSION SEGMENT	BASELINE FLT. RECORDED USAGE
Ascent	12.3%
Descent	14.7%
Cruise	27.9%
Loiter	22.7%
Air-to-Ground	0
TFR	22.4%
	100.0%

The mission mix above produces an n_z plot approximately equal to that of the 500 hours of recorded n_z data.

IX.3.4 Effects of Varying Initial Damage Assumptions

The effects on life and allowable crack growth stress of variation in initial damage assumptions was determined. Initial surface flaw damage levels ranged from $a/Q = .02$ to .125. Initial bolt hole flaw damage levels ranged from .025 to .075 inches.

The results of this study are included in Figures 60 through 62 as plots of maximum allowable spectrum

Table XIII

IMPACT OF USAGE VARIATION
ON LIFE AT CONSTANT STRESS LEVEL

2024-T851 Al. F-111 Baseline Wing
Mid-Point da/dN and K_{Ic} Data

FLAW DESCRIPTION (STRESS LEVEL)	ASSUMED INITIAL FLAW SIZE	LIFE INTERVAL IN FLIGHT HOURS	
		SEVERE USAGE	MILD USAGE
SURFACE FLAW--PART THROUGH	$a/Q = .1$		
$t = .611$, $a/2c = .5$ ($\sigma = 24$ ksi)	$a_o = .246$	2200	10200
SURFACE FLAW--PART THROUGH	$a/Q = .1$		
$t = .611$, $a/2c = .5$ ($\sigma = 25.2$ ksi)*	$a_o = .246$	1500	7250
SURFACE FLAW--PART THROUGH	$a/Q = .1$		
$t = .611$, $a/2c = .5$ ($\sigma = 28$ ksi)	$a_o = .246$	350	1450

* CURRENT BASELINE STRESS LEVEL

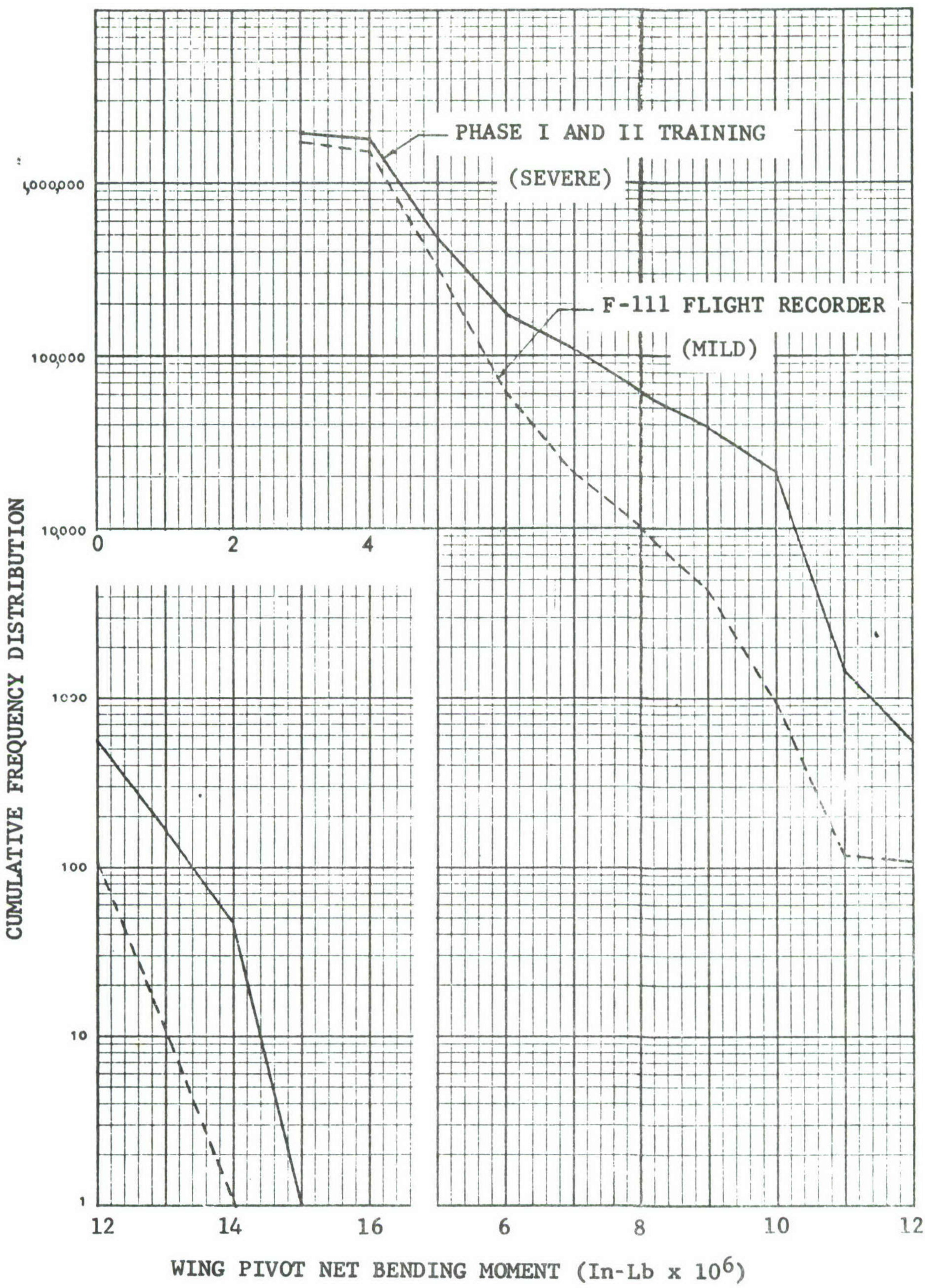


Figure 59 Severe Usage vs Mild Usage Spectra - 4000 Hr. Life

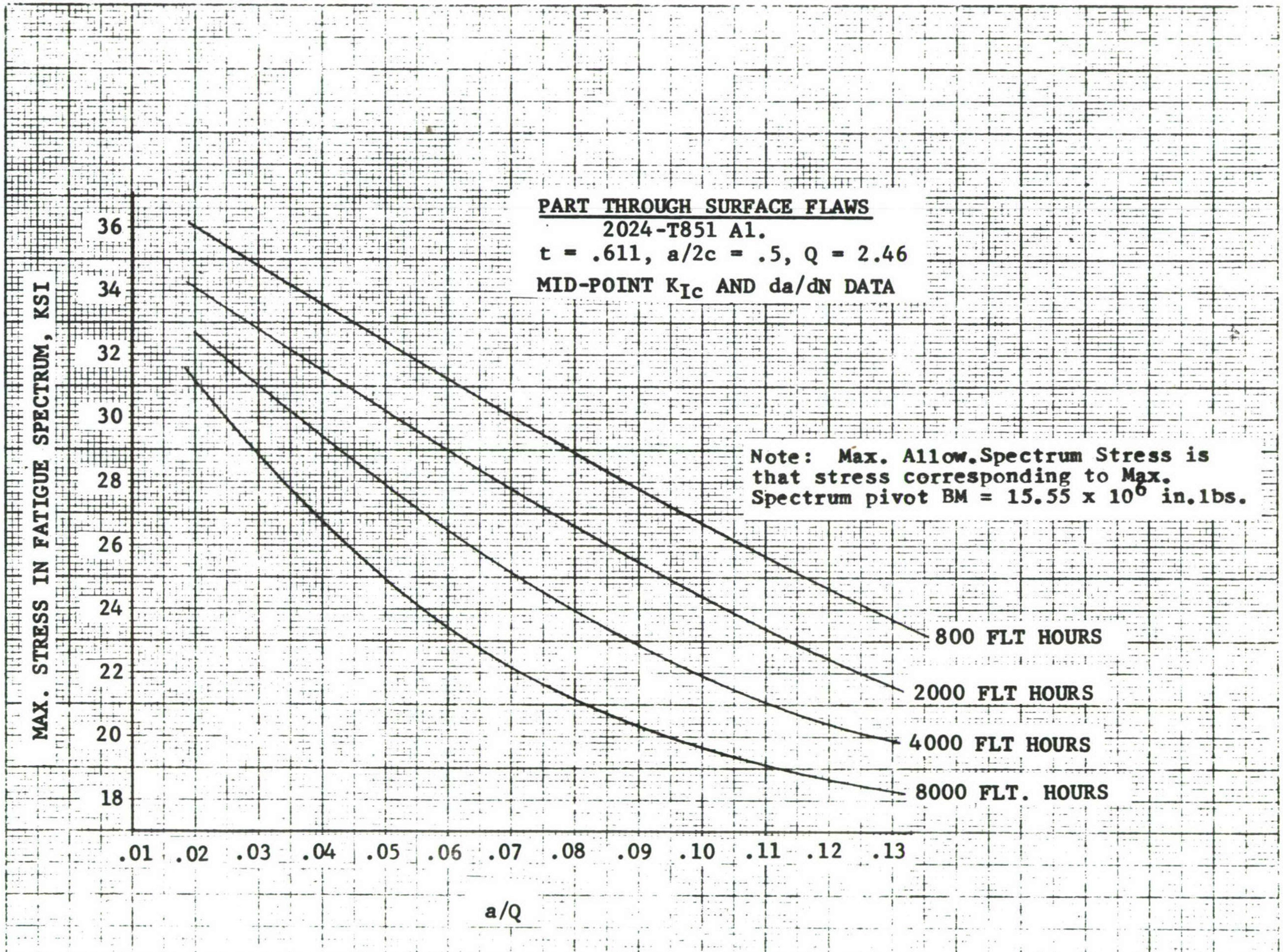


Figure 60 Impact of Initial Flaw Size on Max. Allowable Spectrum Stress

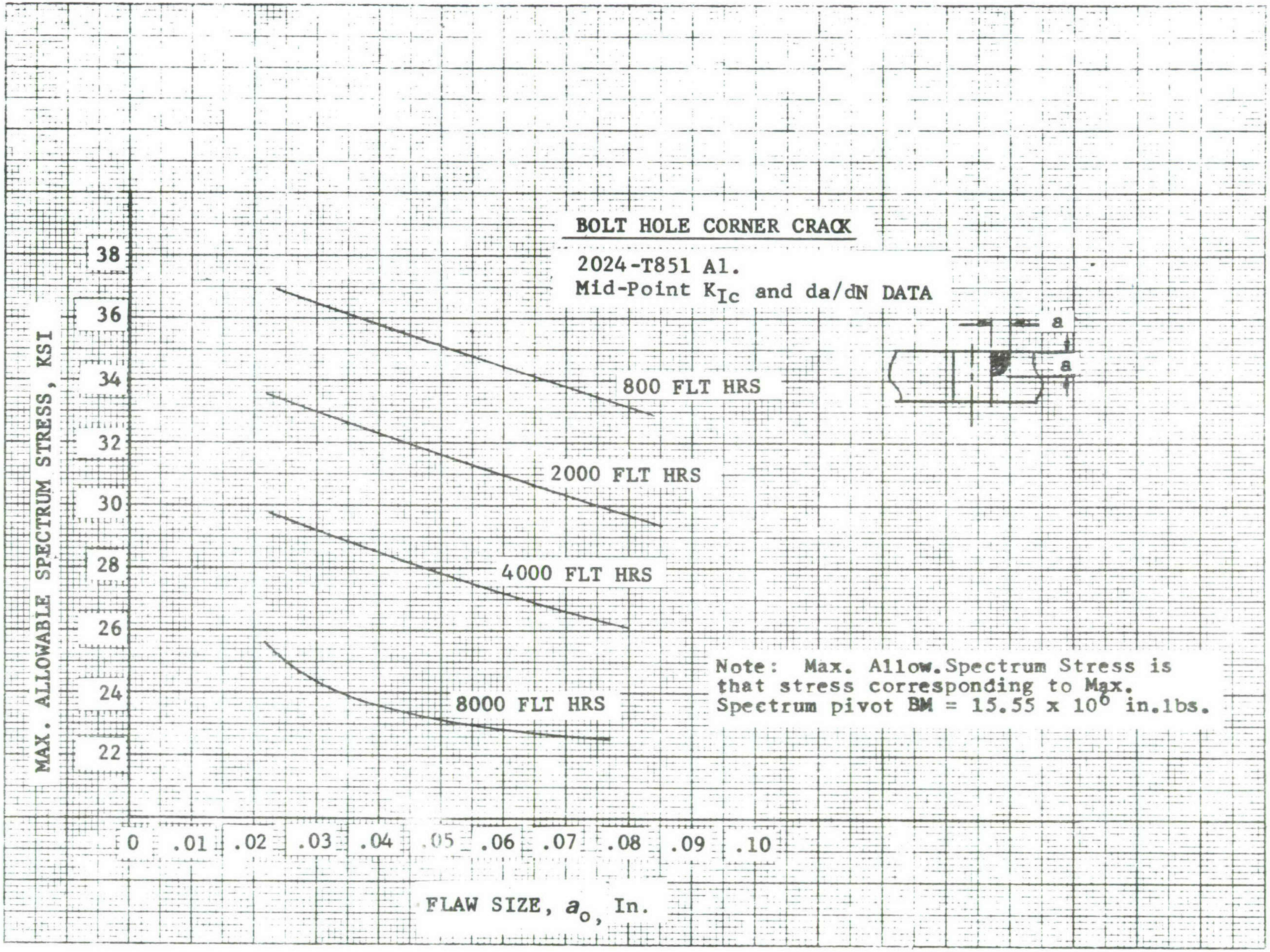


Figure 61 Impact of Initial Flaw Size on Max. Allowable Spectrum Stress

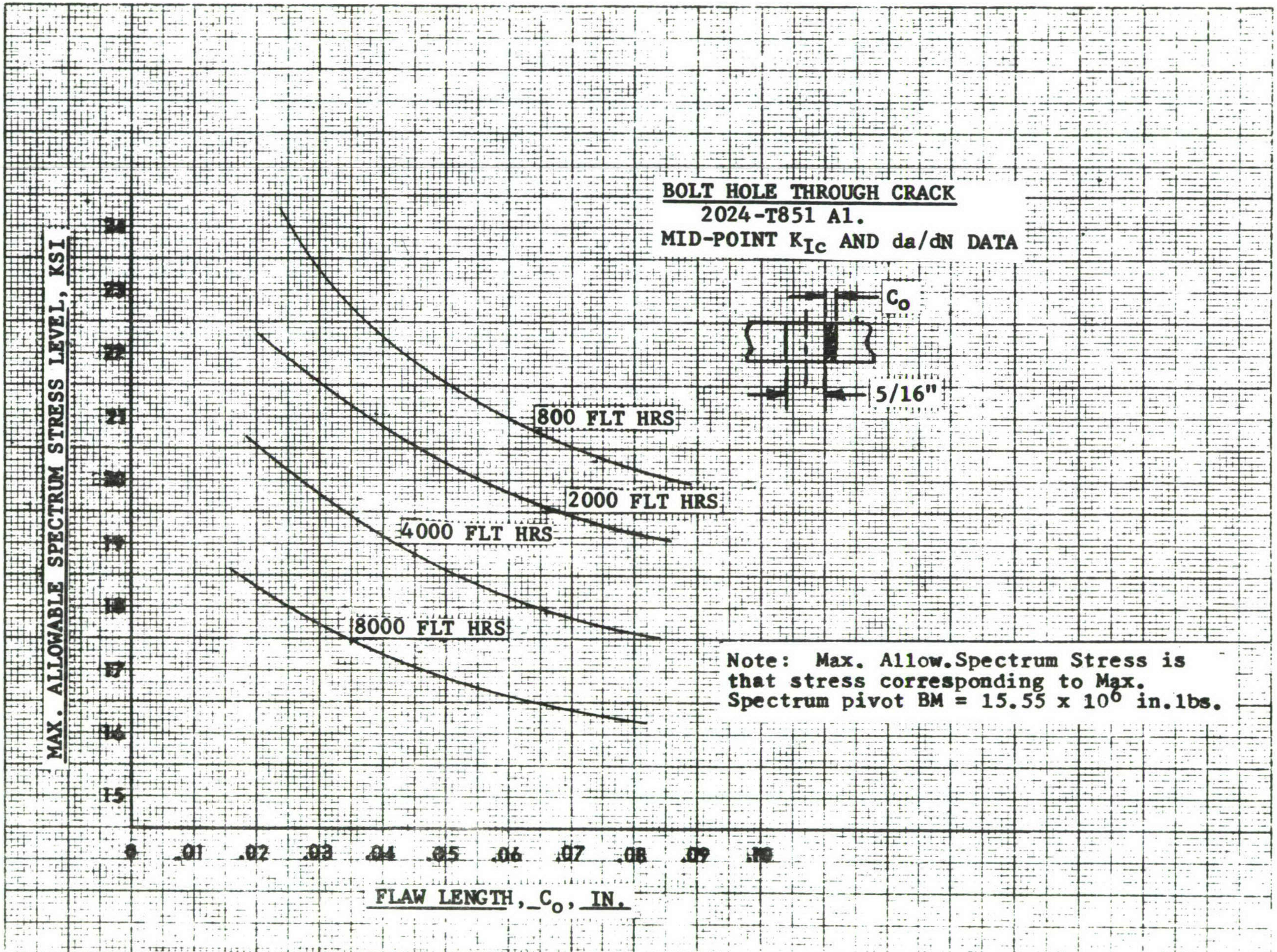


Figure 62 Impact of Initial Flaw Size on Max. Allowable Spectrum Stress

design stress versus initial flaw size for life intervals of 800, 2000, 4000, and 8000 flight hours. The plots were developed using the allowable curves of paragraph IX.3.1 for mid-point fracture data and severe usage. Three flaw types are represented--a part through surface flaw in the .611 inch wing skin, and both corner flaws and through the thickness bolt hole flaws in 5/16 inch diameter holes.

IX.3.5 Baseline Chemical and Thermal Environment Definition

This paragraph requires documentation of the chemical and thermal environment data utilized in F-111 inspection interval analyses.

IX.3.5.1 Baseline (F-111F) Wing Chemical Environment

In conjunction with the F-111 Recovery Program, the possible exposure to corrosive chemical environment was studied according to the plan depicted in Figure 63.

As illustrated in the figure, the basing and flight training requirements were first determined. These requirements were then combined with the attendant climatic data to determine the environmental exposure of the aircraft. Data on the location of critical parts on the aircraft was then established; these data were used in factoring the aircraft environment to determine environmental exposure of the critical parts. The exposure times of the critical parts during the aircraft usage cycle were determined by analysis of the flight training mission requirements.

These environmental studies were directed toward the exposure of critical D6ac steel parts throughout the F-111 airframe. Several of these parts are located in the wing pivot fitting, immediately adjacent to the baseline wing box. Therefore, the chemical exposure data established for the pivot fitting is considered directly applicable to the baseline wing box. Study procedures and results are documented in Convair Aerospace report FZM-12-13249, dated 9 March 1971 (revised 15 November 1972). The completion of this document was originally delayed beyond March 1971 because there was no contractural requirement for its publication after the data became available for use in the inspection interval analyses.

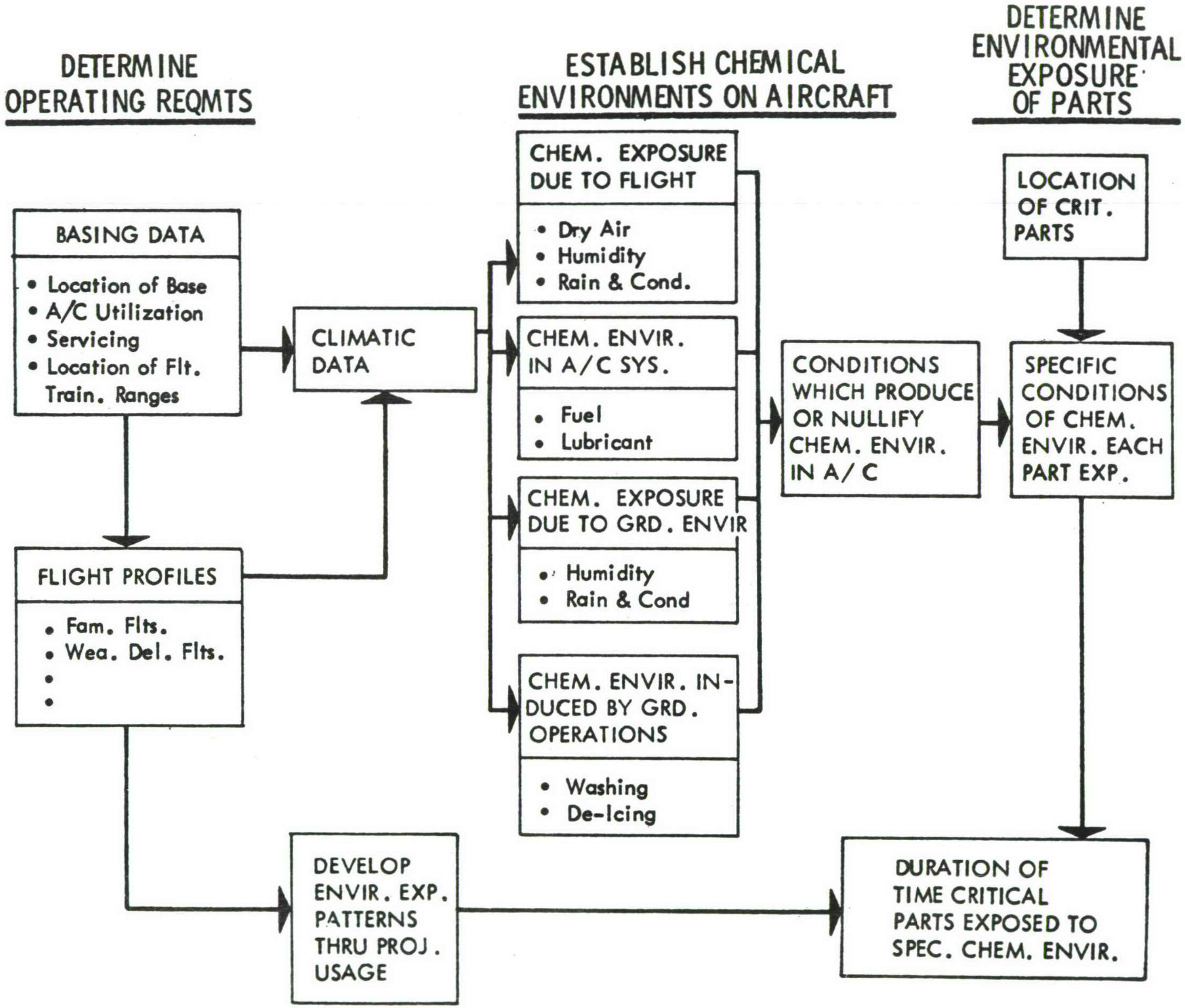


Figure 63 Study Plan of Environmental Exposure Anslyses

Utilization of the F-111 has changed somewhat since this analysis was originally performed (late 1970). The validity of the data in the light of these changes was recently examined. It was concluded that the exposure to chemical environments for current F-111 operations is actually less severe; therefore, the data is conservative. However, the approach described in FZM-12-13249 is considered a unique demonstration of systematically establishing aircraft exposure to corrosive environment.

Basically, four different exposures resulted--dry air, high relative humidity, water (including rain and condensation) and JP-4 fuel. The fuel environment is considered the primary exposure experienced by the baseline wing box fuel tank. While the aircraft are on the ground at their bases, they are normally kept fully fueled in accordance with Air Force operating procedures; therefore, the internal wing "tankage" area is exposed to a fuel environment during this period.

In flight, the fuel management of the F-111 indicates that the fuel usage sequence utilizes the wing tanks first. However, about 10 pounds of unusable fuel remains in each wing. Pressurization and vibration during flight will normally restrict this fuel to the bottom of the tanks, but it is probable that the fuel will randomly immerse all parts in the wing tanks due to motion of the aircraft.

The principal interest in establishing chemical exposures was to determine their effects on flaw growth rates, da/dN and da/dt, and in determining associated K_{ISCC} values. Upper Heyford, England, produced the most significant exposure patterns of those basing locations studied. The exposures for this case are given in Table 2-2 in FZM-12-13249 for both flight and ground conditions. Based on the data for flight conditions (where loadings occur) the following percentages were utilized in performing crack growth analyses for establishing F-111 fleet inspection intervals for critical steel parts:

Fuel tank parts: JP-4 fuel da/dN data 100% of the time.

Other parts: Dry air da/dN 49% of the time, 75--90% relative humidity da/dN data 46% of the time and distilled water da/dN data 5% of the time.

A copy of FZM-12-13249 is included in Section IX.7 in this report.

IX.3.5.2 Baseline (F-111F) Wing Thermal Environment

In conjunction with the F-111 Recovery Program, the exposure to thermal environment was extensively studied to establish minimum structural temperatures for critical D6ac steel parts throughout the airframe. Several of the parts studied are located in the wing pivot fitting which is immediately adjacent to the baseline wing box. Therefore, the approach and results of the Recovery Program thermal exposure studies are directly applicable to the baseline wing box.

The thermal history during flight is of primary interest relative to the calculation of critical flaw size for service operations. Critical flaw size is directly related to the square of the value of K_{Ic}, and the K_{Ic} of D6ac steel is lower for lower temperatures. Consequently, smaller critical flaw sizes may be experienced during in-flight operations. The minimum structural temperature established for use in inspection interval analyses of D6ac steel parts for the F-111 was + 10°F which corresponds to minimum temperature for standard day operations. It should be noted that the fracture toughness of 2024-T851 aluminum alloy is not significantly sensitive to temperature variations as shown by the data in Table XIV. However, definition of thermal environment is an important element of fracture control and the system used during the F-111 program is a good example of a proven approach. A brief description of the Recovery Program study is given in the paragraphs below.

Structural temperatures on the F-111 were measured by NASA during seven flights at Edwards AFB in 1967. Approximately 60 thermocouples were active during these flights. Flight conditions were high subsonic to low supersonic speeds at altitudes from 20,000 to 50,000 ft., with ambient temperatures near those of a tropical day. Flight durations were typically 0.75 to 1.5 hours. The lowest temperature recorded was + 26°F on a vertical tail skin panel. The lowest pivot fitting temperature was + 37°F. Structural temperatures were measured on other airplanes, but always to investigate high temperature problems. The primary conclusions were (1) that the available flight test data does not provide extensive enough information on minimum structural temperatures, and therefore, (2) the minimum temperatures must be based on analytical predictions.

Table XIV

EFFECT OF TEMPERATURE ON K_{Ic} FOR 2024/2124-T851 ALUMINUM ALLOY

Neither cold temperatures down to -65°F nor short time elevated temperature exposure at temperatures up to 300°F effect the K_{Ic} value for 2024-T851 or its higher purity version 2124-T851 aluminum alloy plate. This is evidenced by the data shown in the table below. Ranges must be compared because of data scatter. Specimen orientation shown is that of MCIC-HB-01.

Alloy and Condition	Form and Thick In.	Test Temp °F	No. Spec.	Spec Design	Spec Orient	Spec. Dim B In	Spec. Dim W In	Range K_{Ic} (KSI SQ.IN.)	Ref.
2124-T851	P 4.25	70	3	CT	T-L	1.0	2.0	24.4-25.0	(1)
	P 4.25	-40	3	CT	T-L	1.0	2.0	24.8-25.4	(1)
	P 4.25	70	3	CT	S-L	1.0	2.0	21.0-22.8	(1)
	P 4.25	-40	3	CT	S-L	1.0	2.0	21.3-22.0	(1)
2024-T851	P 1.39	70	3	BEND	T-L	1.38	3.0	20.1-20.5	(2)
		-112	2	BEND	T-L	1.38	3.0	21.3-22.7	(2)
		-320	2	BEND	T-L	1.38	3.0	22.1-22.2	(2)
	P 3.0	-65	3	CT	L-T	.75	1.5	24.4-27.6	(3)
		0	3	CT	L-T	.75	1.5	25.2-29.5	(3)
		70	2	CT	L-T	.75	1.5	26.9-27.3	(3)
		200	3	CT	L-T	.75	1.5	25.9-27.8	(3)
		300	3	CT	L-T	.75	1.5	26.6-27.3	(3)
	P 3.0	-65	1	CT	T-L	.75	1.5	23.3	(3)
		0	2	CT	T-L	.75	1.5	20.8-22.9	(3)
		70	3	CT	T-L	.75	1.5	19.7-22.6	(3)
		200	3	CT	T-L	.75	1.5	20.7-22.8	(3)
		300	2	CT	T-L	.75	1.5	21.7-22.3	(3)
	P 3.0	-65	3	CT	L-S	.75	1.5	29.3-31.4	(3)
		0	2	CT	L-S	.75	1.5	31.4-31.7	(3)
		70	3	CT	L-S	.75	1.5	30.0-32.2	(3)
		200	2	CT	L-S	.75	1.5	29.7-30.9	(3)

References:

(1) Convair Fort Worth Tests

(2) MCIC-HB-01, Reference 84288

(3) MCIC-HB-01, Reference 83243

An extensive thermal analysis was therefore performed to determine the minimum temperatures of D6ac steel parts as installed and operated on the F-111 airplane. Transient temperature distributions within the steel components were computed for representative mission profiles and atmospheres. Good agreement was obtained between computed temperatures and limited flight test data which provides verification of the calculations. The results of this study define : (1) the steel parts and corresponding aircraft locations, (2) the representative mission profiles considered, (3) the atmospheric conditions, and (4) the resulting structural temperature histories. The results are contained in Section IX.7.

The primary conclusion of this study was that the temperature of the D6ac steel components will not be less than + 10°F during flight, on a standard day, at a time when appreciable structural loading will occur. The most obvious example of a potentially lower (less than + 10°F) structural temperature will occur during a ferry mission at M = 0.75 at altitudes around 30,000 ft., on a standard day; although the structural temperature is -6°F, the structural loading is negligible at such flight conditions. Structural temperatures associated with a polar day will approach -15°F. However, the likelihood of maximum load occurring at a polar day minimum temperature is considered extremely remote for the F-111 based on planned usage and operating limits.

IX.3.6 Review of Manufacturing NDI

Baseline wing manufacturing NDI experience has been reviewed, including use and applicability of the proof test concept. The review included operations beginning with the first metal cutting step and ended with final sell off of the aircraft at Fort Worth. The type, frequency, and results of nondestructive inspection are documented in this section.

The proof test philosophy as applied to the F-111 wing and subsequent use of proof test results in F-111 inspection interval analysis is discussed in paragraph IX.3.6.3.

IX.3.6.1 Manufacturing NDI Review

The review was limited to details and assembly of the wing box proper. The wing pivot assembly, flight control structure, and pylon housings were not included.

The following part numbers and nomenclature identify the considered detail parts:

12W950	Wing skin, upper	12W915	Bulkhead #5
12W951	Wing skin, lower	12W914	Bulkhead #4
		12W926	Bulkhead #3
12W908	Front spar	12W919	Bulkhead # 3.5
12W902	Fwd aux. spar	12W918	Bulkhead # 2.5
12W903	Center spar		
12W904	Aft aux. spar	12W821	Bulkhead, Outer housing
12W905	Rear spar	12W820	Bulkhead, Inner housing
		12W912	Bulkhead # 2
12W972	Doubler	12W917	Bulkhead # 1.7
12W974	Doubler	12W823	Bulkhead, Pylon housing
12W986	Doubler		
12W973	Doubler	12W824	Pylon housing support
12W982	Splice	12W822	Bulkhead
12W985	Web spar	12W920	Bulkhead # 1.5
12W983	Flange	12W911	Bulkhead # 1.0
12W988	Splice	12W916	Bulkhead # 0

Each of the detail parts, aside from dimensional inspections, received one penetrant inspection per NDTS 10.00, Liquid Penetrant Inspection. Each part, except 12W974 (titanium) received one hardness test per NDTS 15.00, Hardness Testing, Method of Inspection. The wing spar raw material received an ultrasonic inspection per NDTS 50.00 Ultrasonic Inspection, Method of. At assembly the wing box structure received a radiographic inspection per NDTS 30.00-7, X-ray inspection of F-111 Wing. No further NDI's are accomplished in system operations or cold proof test, although a visual examination is conducted on the exterior of the wing skins after proof test.

Copies of the referenced NDTS's are included in Report MEA-301, included in Section IX.7 in this report.

Coordination with Manufacturing Engineering revealed that within each nomenclature family the manufacturing processes experienced were the same; therefore, one representative part was selected from each nomenclature family. They were:

12W985-9/-10	Web spar	12W908-25/-26	Front spar
12W951-9/-10	Lower wing Skin	12W974-9/-10	Doubler
12W915-15/-16	Bulkhead		

In addition, the assembly of the wing box, Items 62, 61, and 60 was reviewed along with 12AEI-11-1047B, Cold Temperature Proof Load Test of F-111F Aircraft.

Figures 64 through 69 illustrate the principle manufacturing steps and inspections performed on the parts from fabrication to delivery. The NDTS used and its sequence is also shown. Table XV summarizes the NDI experience. The proof test sequence is illustrated in Figure 70.

IX.3.6.2 <u>Results of Manufacturing NDI</u>

The manufacturing history was obtained from a special computer report run against the Quality Data Records for the parts listed above. Two discrepancy request codes were utilized, "190" for Quality Assurance Reports (QAD's) and "PACRK" for Discrepancy Reports (DR's). The report covers only the wing box structure detail reports and wing-box structure assembly. It did not cover the pivot assembly. The part involved, crack location, and crack size is summarized in Table XVI. Figure 72 shows part location within the overall wing.

IX.3.6.3 <u>Proof Test of Aircraft</u>

The proof test concept has been used for many years to provide assurance of in-flight safety in the missile and rocket field. Proof testing of fleet aircraft was recently initiated as part of the F-111 Recovery Program.

A basic objective of the F-111 proof test program is to screen the structural system for gross defects including material flaws and any other defect not amenable to standard inspection practices, such as improperly seated bolts, steel parts with improperly heat treated areas, etc. Whereas the static and fatigue testing of full-scale airframes evaluate the design integrity, the proof test of each production airplane supplements the nondestructive inspection procedures and proves to a given level the integrity of the manufacturing process for that individual aircraft. Quality control and inspection of materials, component parts, and assembled airframe, using accepted standard practices, is subject to variation depending on many variables. This is not to say that quality assurance programs are ineffective, quite the contrary. However, the probability that one or more defects will remain undetected in manufacturing a fleet of aircraft is finite. The proof

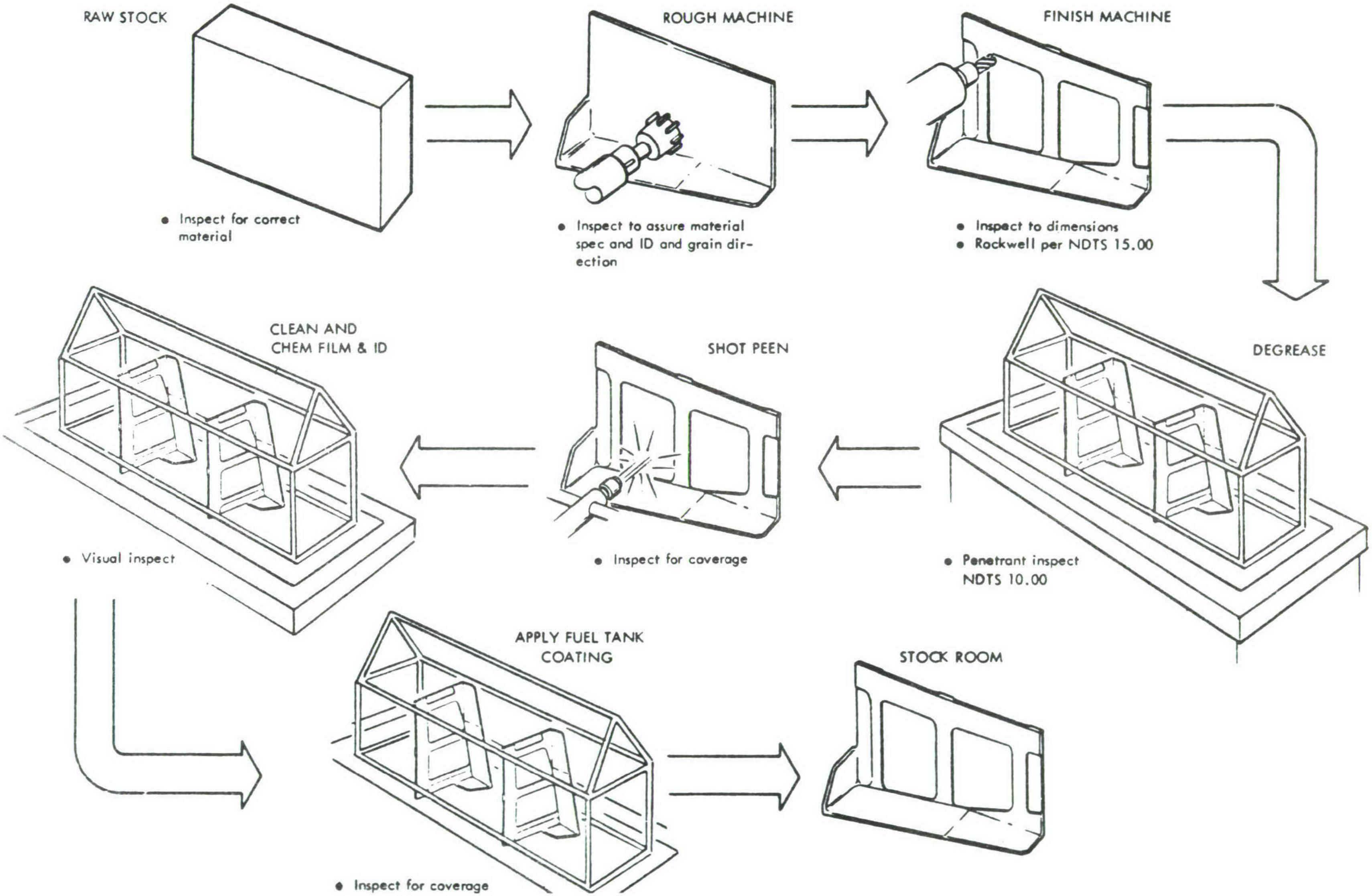


Figure 64 12W985-9/-10 Web Spar, 2024-T851 Material (FMS1010-T851)

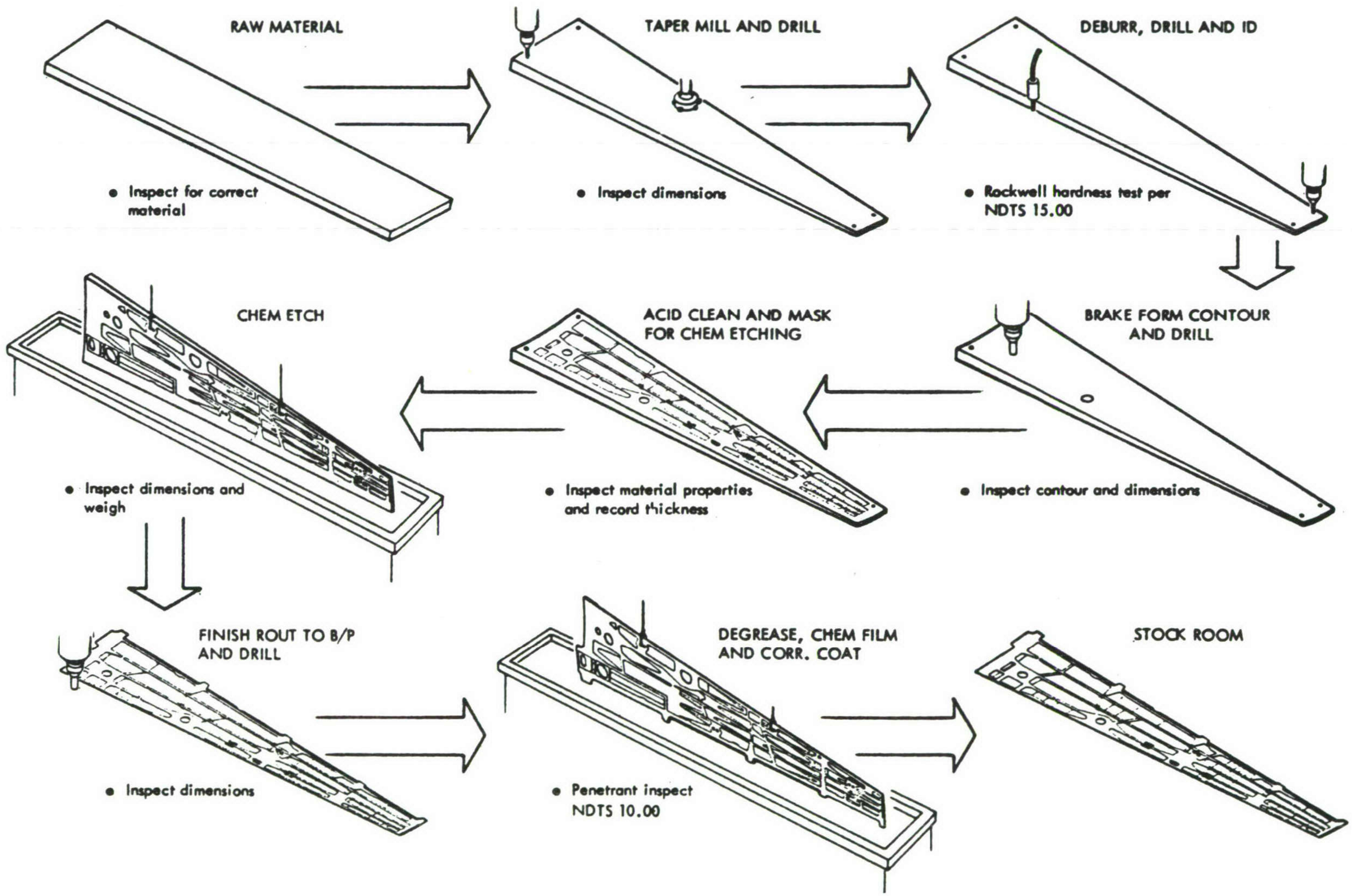


Figure 65 12W951-9/-10 Lower Wing Skin, 2024 Al Plate (QQA 355-T351)

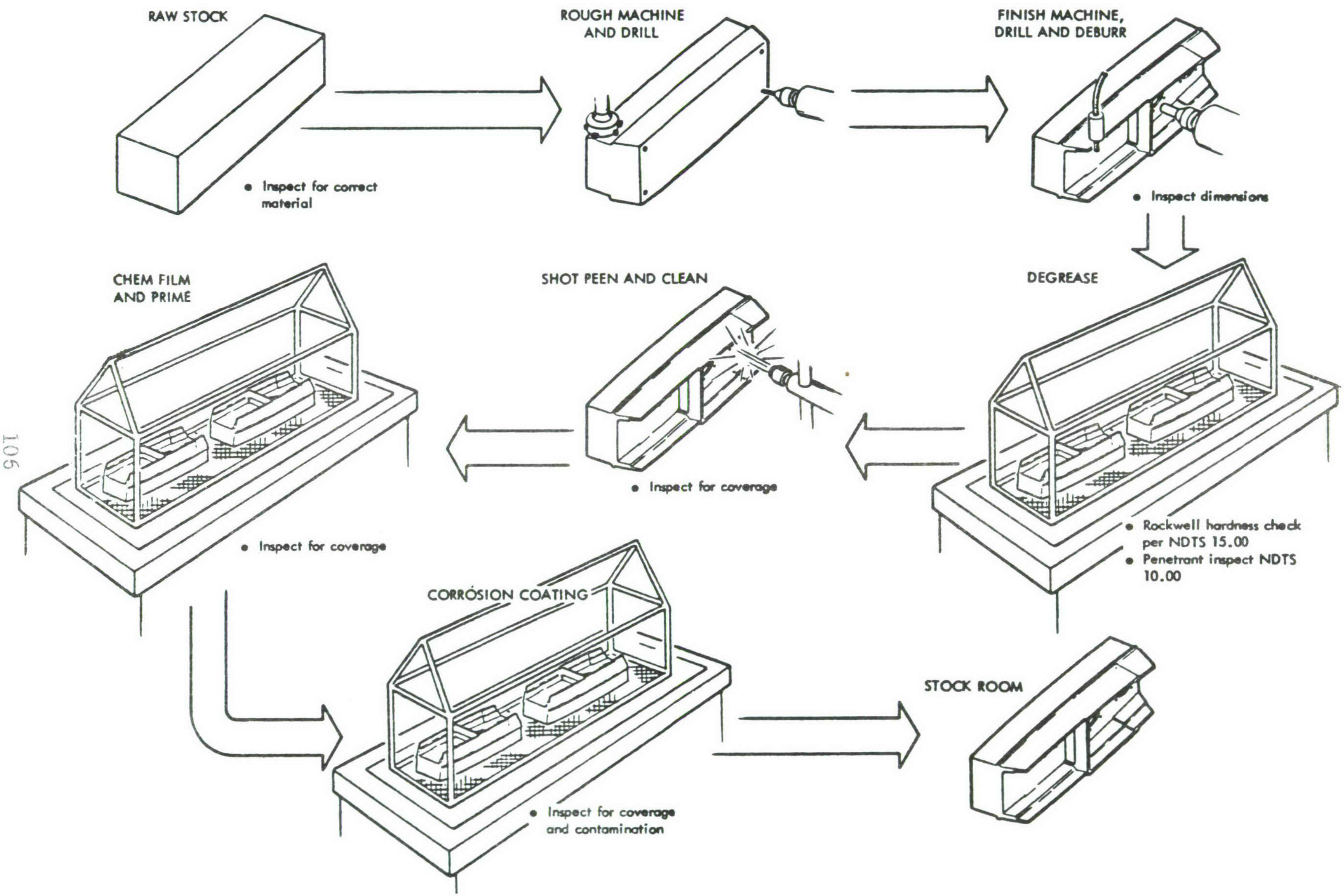


Figure 66 12W915-15/-16 Bulkhead #5
2024-T851 A1 (FMS 1010)

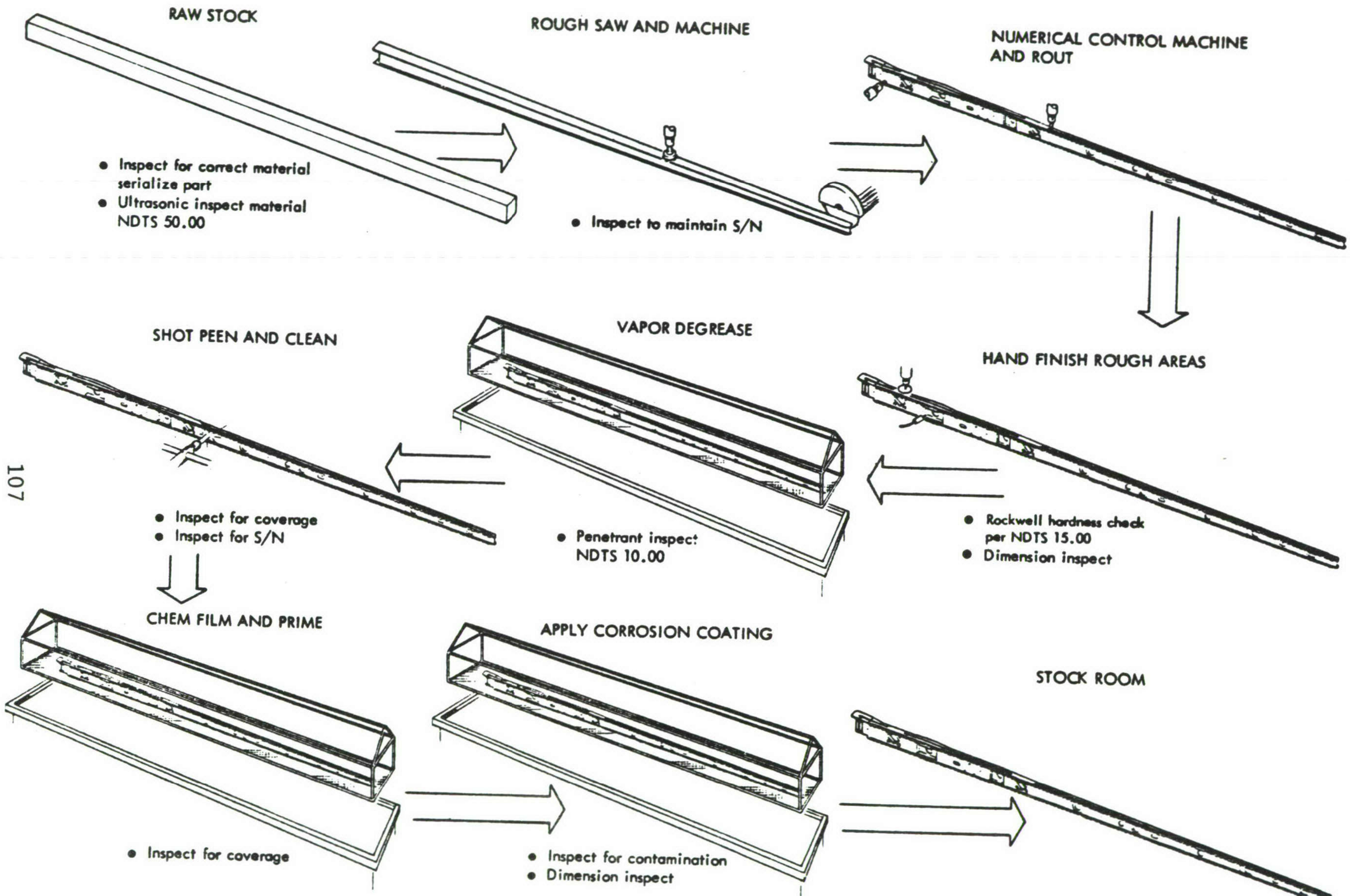


Figure 67 12W908-25/-26 Front Spar, 2024-T851 Al (FMS 1010 Flat)

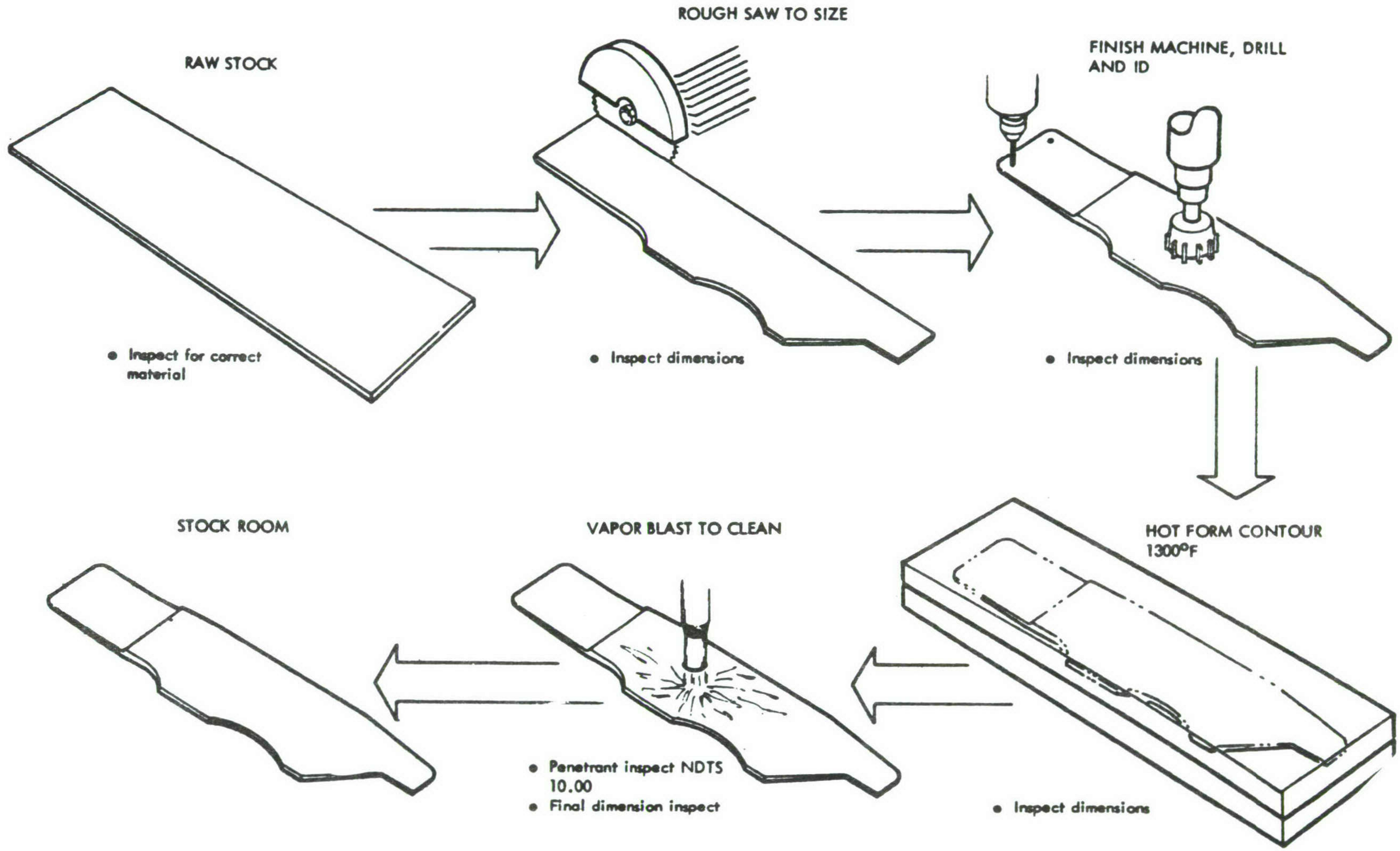


Figure 68 12W974-9/-10 Doubler, 6Al-4V Titanium (MIL-S-9046, Class #2)

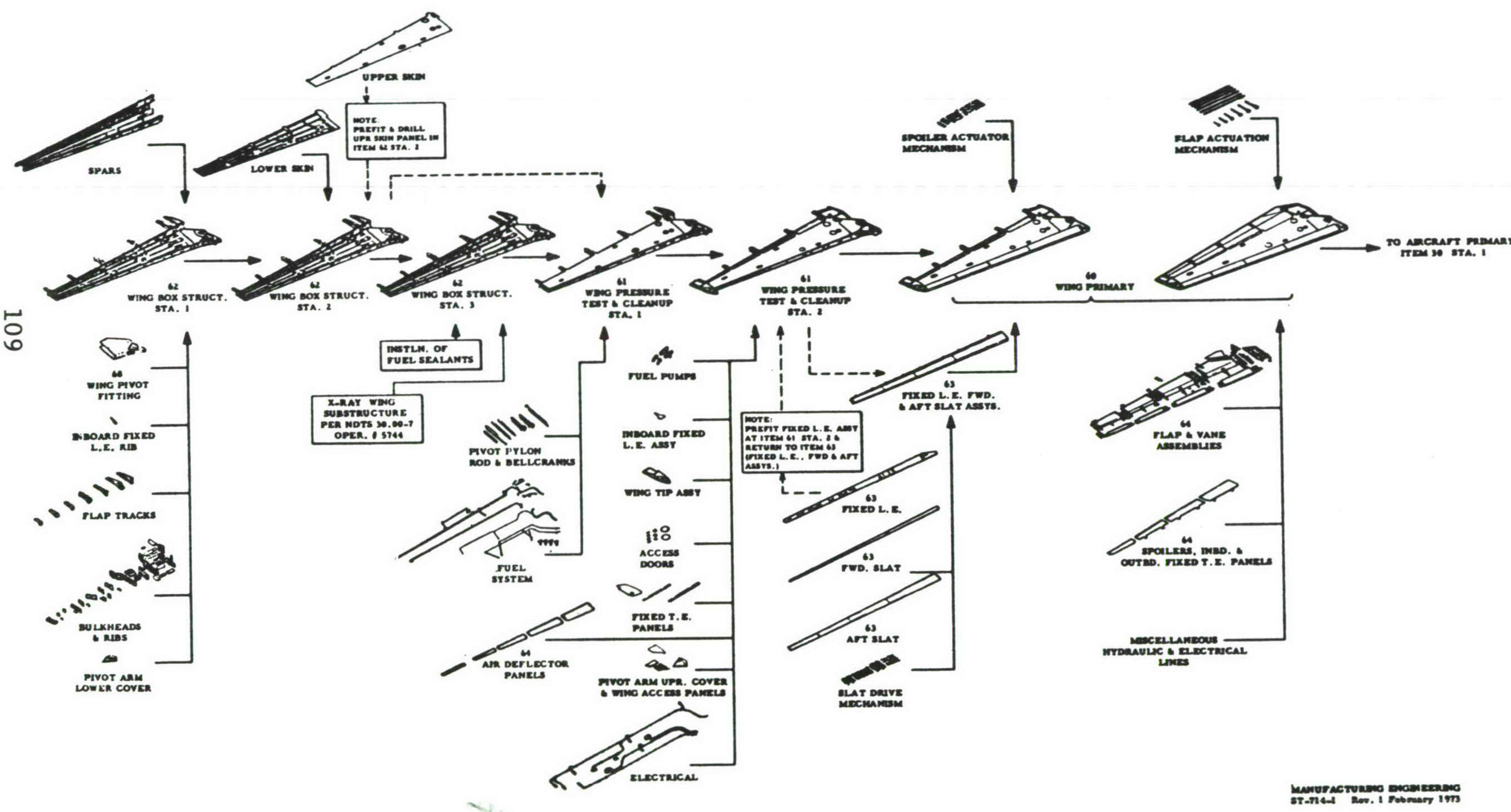


Figure 69 F-111F Wing Manufacturing Sequence

Table XV WING BOX STRUCTURE NDI EXPERIENCE SUMMARY

NDI PERFORMED	12W985 WEB SPAR	12W951 WING SKIN	12W915 BULKHEAD	12W908 WING SPAR	12W974 DOUBLER	ASSEMBLY WING BOX
Ultrasonic per NDTS 50.00 (3)				1 Time		
Rockwell Hardness per NDTS 15.00	1 Time	1 Time	1 Time	1 Time		
Penetrant Inspection per NDTS 10.00 (1)	1 Time	1 Time	1 Time	1 Time	1 Time	(4)
X-ray Inspection per NDTS 30.00-7 (2)						1 Time

NOTE:

(1) No formal minimum flaw size established for F-111.

(2) See Figure 71 for X-ray locations.

(3) For flaw size criteria, see NDTS 50.00. (Dependent upon class of flaw)

(4) Each tapered hole receives an airgage inspection for roundness and a visual inspection for finish and cracks. Dye penetrant is used on suspicious holes to finalize determination. Straight holes receive a visual/dimensional inspection with dye penetrant NDI used to resolve questionable areas.

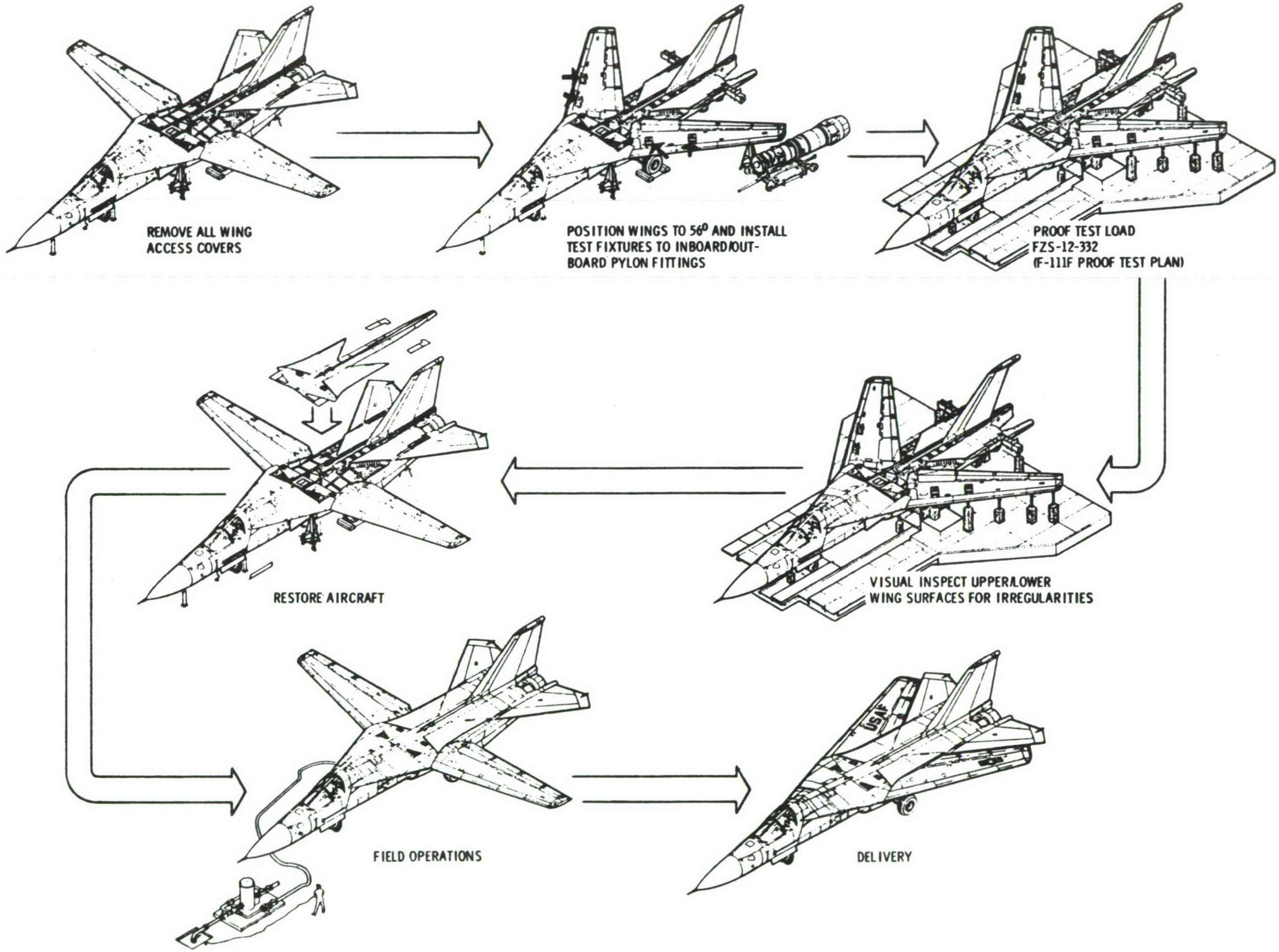
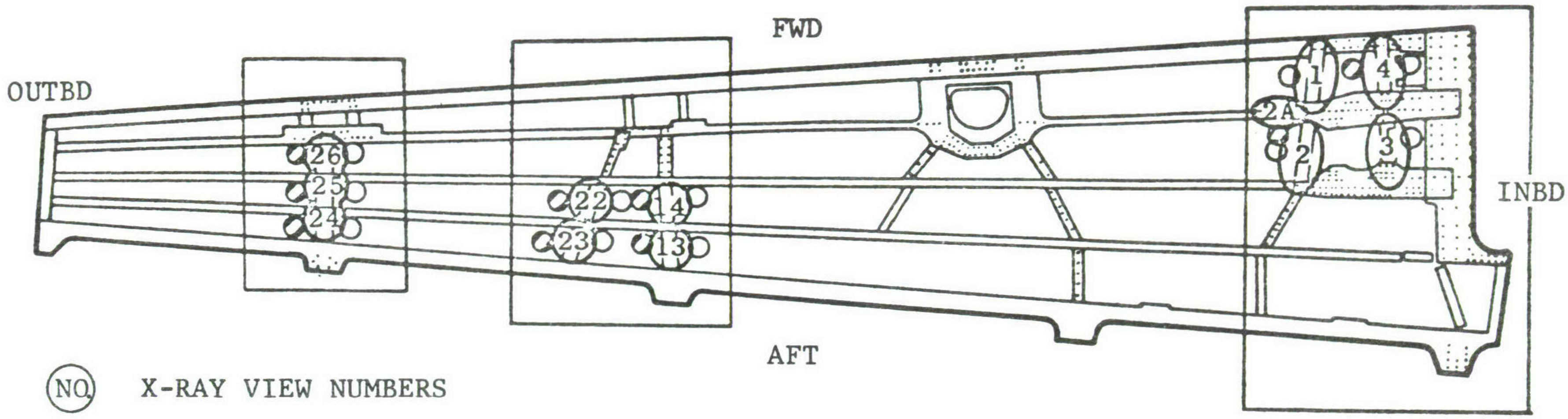


Figure 70 F-111F Aircraft Wing Proof Test, (Cold Temperature) 12AEI-11-1047B

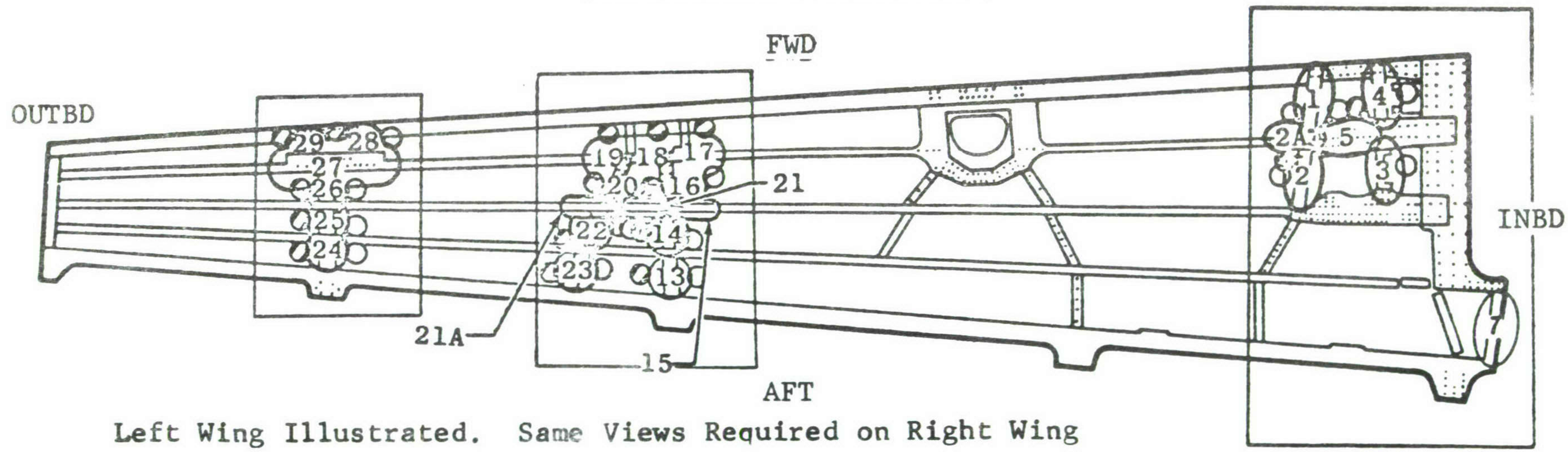


(NO) X-RAY VIEW NUMBERS

○ = FOCAL SPOT PLACEMENT, REGULAR SHOT

⊘ = FOCAL SPOT PLACEMENT, "A" SHOT

LEFT/HAND BOTTOM VIEWS



Left Wing Illustrated. Same Views Required on Right Wing

Figure 71 X-Ray Locations (Ref. Table XVI)

Table XVI

F-111 WING BOX STRUCTURE CRACK HISTORY SUMMARY
16 AUGUST 1969 TO 1 JANUARY 1973
(PRODUCTION AND MODIFICATION)

RADIOGRAPHIC DATA (NDTS 23.04)

Part Number	No. of Cracks	Crack Locations	Min/Max Size
12W903	2	X-Ray View #34 of NDTS and between hole #232 and hole #214	.50 Lg - 1.50 Lg
12W914	2	X-Ray View #24 and X-Ray View #25 of NDTS	.50 Lg - 1.10 Lg
12W920	9	X-Ray View #33 (4 times), Hole #175, Hole #173, X-Ray View #V-31, X-Ray View #V-31A, and X-Ray View 30-30A of NDTS	.3 Lg - .62 Lg
12W027 Assy	6	Hole #179, Hole #178, X-Ray View 14A, X-Ray View #4A, at .50 dia hole, and at rear spar lug at bolt hole	.5 Lg - 1.25 Lg

LIQUID PENETRANT DATA (NDTS 10.00)

Part Number	No. of Cracks	Crack Locations	Min/Max Size
12W902	1	B/P Zone 75-C, Sht #7, in radius	2.50 Lg
12W903	2	B/P Zone 4B (2 times)	.50 Lg (2 times)
12W905	2	At .199 hole & Sta 353 on integral lug	.050 Lg
12W855	2	At Sta. 164 (runs into hole) and Sta. 82 (runs into rivet hole)	.10 Lg - 1.45 Lg
12W908	8	B/P Zone 45B, B/P Zone 46 B/C, B/P Zone 43B, B/P Sht #3, Sec 37C, at 226.608 dim line of B/P, Sta. 215, Sta. 318.112, on L/S tab, Sta. 87.25 (runs into rivet hole)	.10 Lg - 1.45 Lg
12W914	1	B/P Zone 4B, Sec G-G	1.2 Lg
12W916	1	Sta. 87.25 (runs into rivet hole from edge)	.10 Lg
12W926	2	B/P Zone 5B	1.25 Lg
12W920	11	Between front aux. spar & front spar in splice area (2 times), upper side of cap, Hole #177 (lwr surface), Fwd edge of cap, On 12W896 cap, Lwr surface next to .65 dia hole, upper side of cap, Hole #180, and Hole #173.	.10 Lg - 2.25 Lg

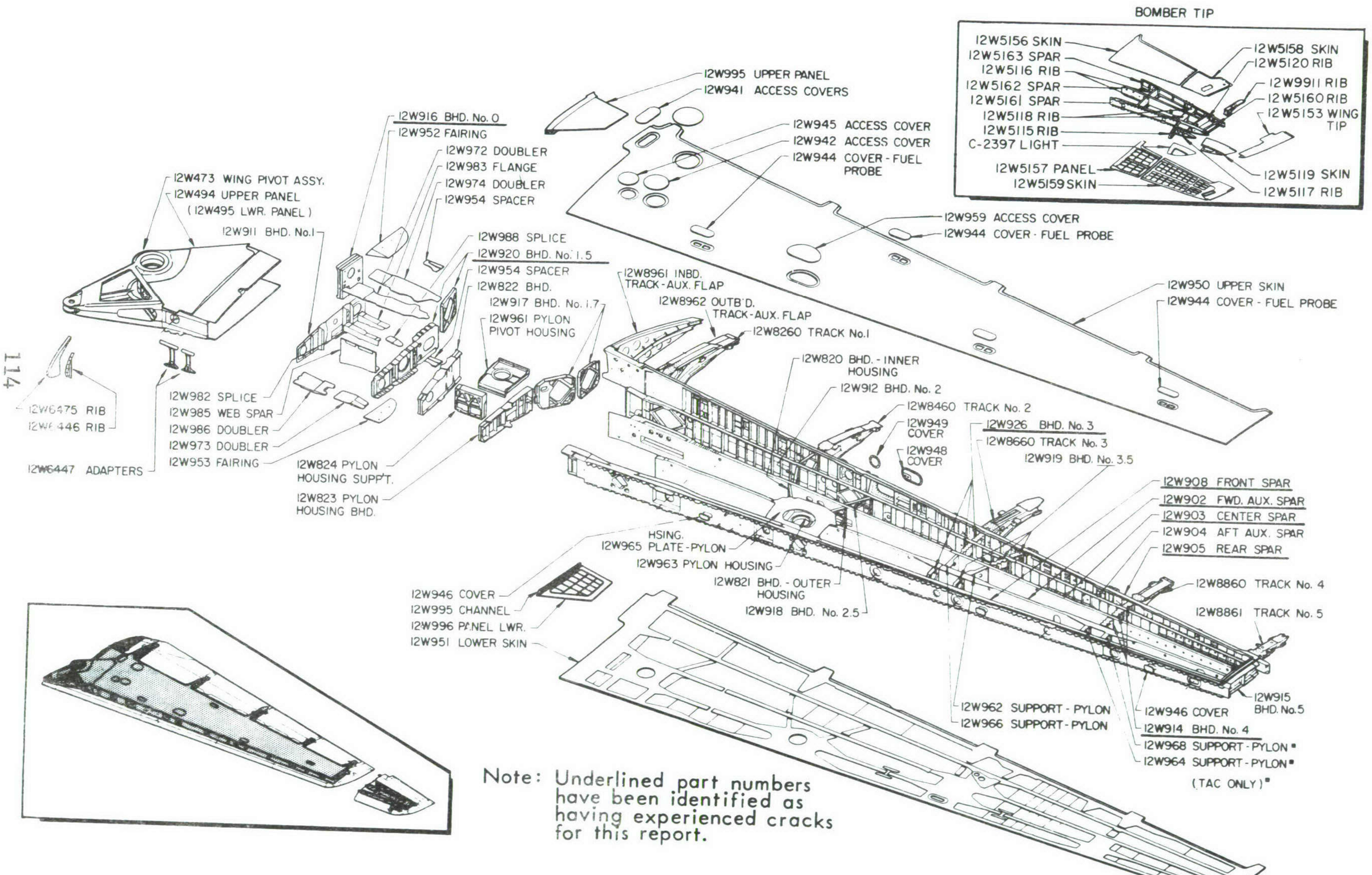


Figure 72 F-111 Wing Box Exploded View

test provides positive assurance that static strength capability of every airplane has not been impaired at least to the proof level even though a defect might exist. In this role, the proof test is an invaluable inspection tool.

A second objective of the proof test program is to provide a basis for establishing inspection intervals for use with the fleet in service. By definition, an inspection interval is that period of time safe for flight operations assuming that a material flaw is either present in the structure initially or develops in service. The interval is determined by establishing the following:

(1) The initial size (a_i) of the assumed flaw based on the conditions of the proof test.

(2) The growth of this flaw as a function of time under the service conditions of load and environment.

(3) The critical size (a_c) of the flaw for the service operations.

The inspection interval is then determined as the growth interval to failure divided by an appropriate confidence factor. The concept for inspection interval determination is illustrated in Figure 73.

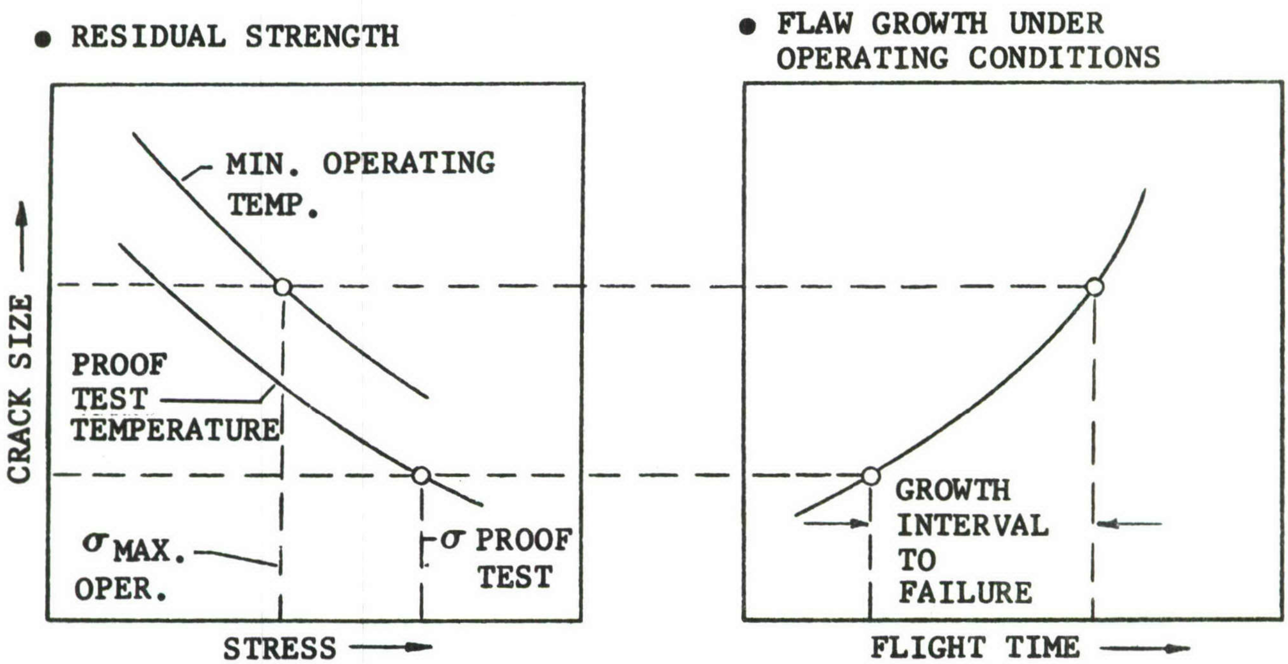


Figure 73 Proof Test Concept for Determining Inspection Intervals

The case shown is applicable to materials whose residual strength with a given size flaw present decreases with temperature.

IX.3.6.3.1 Calculating the Inspection Interval. A number of basic elements are included in the approach used to establish the inspection interval. These elements are related as shown in Figure 74.

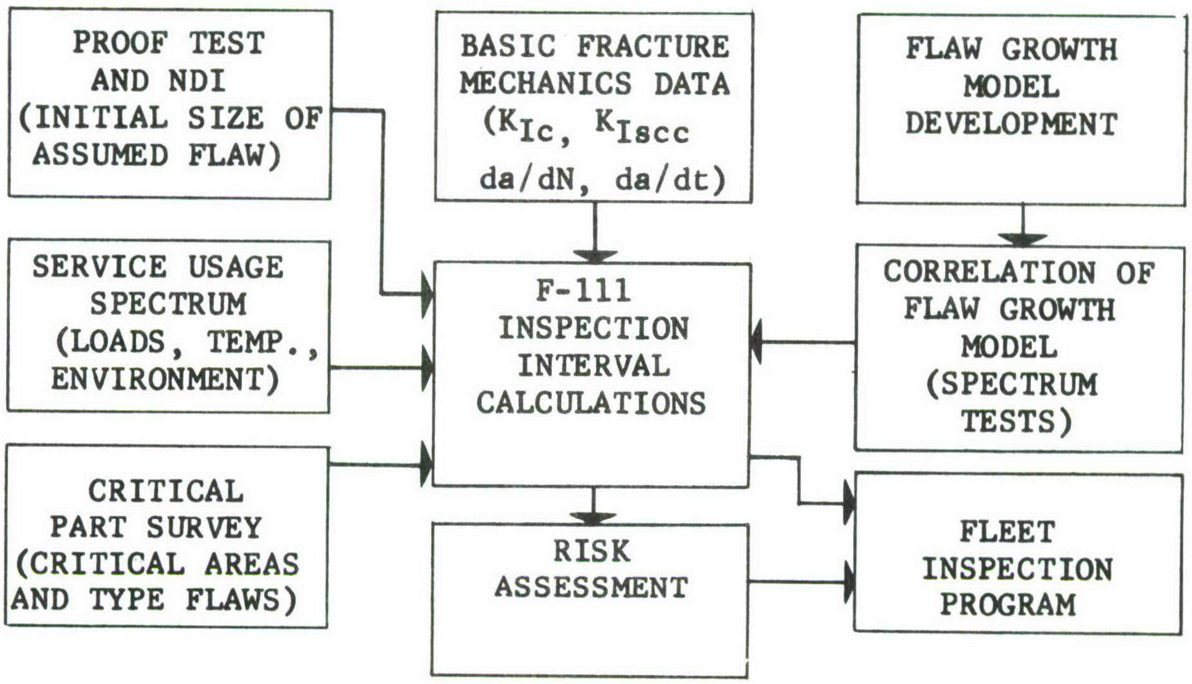


Figure 74 Inspection Interval Calculations

The two basic objectives of the proof test as discussed above do not seem to be a viable option when considered for application to the ADP program. As stated previously, the proof test concept shown in Figure 73 is applicable only to materials whose residual strength with flaws present decreases with temperature. The 2024-T851 aluminum used in the baseline, and most other aluminums (and titanium) do not exhibit this characteristic. In addition, the residual strength requirements in the current and proposed damage tolerance criteria are at (or near) limit load, and proof test loads are also generally specified as limit load. This was the case with the F-111 proof test program.

IX.3.7 NDI Demonstration Program

Providing an optimum structural design requires a balance between minimizing weight and maximizing structural integrity. For example, it was shown in paragraph IX.3.1 that designing the baseline to an allowable flaw size requirement of 0.050 inch (bolt hole flaw) resulted in a weight increase of 205 pounds per wing. If the required flaw size could be reduced to 0.025 inch by an NDI demonstration program, the minimum allowable spectrum stress from Figure 26 could be increased from 16.9 ksi to 18 ksi.

Entering Figure 40 indicates that the wing will weigh about 1725 pounds, or 175 pounds over the current baseline weight of 1550 pounds. Thus the demonstration program would save 30 pounds per wing (60 pounds per A/C) if it could demonstrate that an 0.025 crack could be detected 90% of the time with a 95% confidence level.

A typical demonstration program is discussed in the following sections.

IX.3.7.1 Variables Affecting Demonstration Program

In establishing an NDI demonstration program, several variables will impact the cost and the accuracy of the results. These include configuration of the specimen used (flat, simple geometry, or complex), number and kinds of defects and methods for minimizing operator awareness. These are discussed in more detail in the following paragraphs.

IX.3.7.1.1 Inspector Awareness/Anticipation Factor. Based on CA/FW's previous Human Factors Evaluations (ECP 10544, NDI Improvement Program), several conclusions based on numerous observations can be recognized.

a. To an inspector who handles the same parts, it is difficult to disguise a test specimen regardless of the amount of effort expended. The least bit of deviation from routine will alert the inspector. It can be minimized, however, by careful preparation of planning documents, specifications and adequate supervision in the test area.

b. Inspectors will react differently to the program. One may become nervous and lose efficiency. Another may intensify his efforts beyond a normal evaluation.

c. Any variation in equipment such as holding fixtures, probes, etc., may also distort a normalized approach.

d. Based on CA/FW's previous human factors experience, if a sufficient number of inspectors and a sufficient number of tests (approximately 150 in lieu of 31) are utilized, the total overall effect of inspector awareness/anticipation is minimized to a point within a normal expected variation in achievement. Thus, the summarized results from an extensive evaluation program in which inspector awareness is considered should reflect a norm equivalent to routine production.

IX.3.7.1.2 Advantages/Disadvantages of Simple (Flat Plate) Demonstration Program versus Complex Demonstration Program (Production Configuration). A simple demonstration program is obviously advantageous with regard to cost. Test specimen, fatigue, tooling, processing time, etc., are greatly reduced in terms of expense, but in reality, the specimens utilized in a simple program do not reflect the configuration of a production part. Basically, the inspection of a flat plate configuration reflects the capability and performance of the NDI equipment considerably more so than the NDI technique and/or inspector. However, for many detail parts, machined, or formed, or welded, the flat plate approach is the most cost effective and is adequate.

A production structure demonstration program is advantageous with regard that the results are directly translatable into design criteria. In addition, inspector awareness/anticipation is minimized due to its visually familiar appearance. However, the cost to such a program is increased due to specimen manufacturing, crack inducement/control, and demonstration effort.

A significant advantage lies in the verification of the reliability of the NDI technique to successfully operate in a production environment. This is a point often overlooked in NDI demonstration programs. Most NDI techniques are developed in a laboratory and demonstrated on flat plates, then applied to production with a minimum of intelligence on the NDI's reliability when applied to a production effort.

The degree of increased difficulty in applying the NDI technique on production structure prohibits the lowering of the number of tests required that otherwise would occur (in comparison to flat plates) by minimizing inspector awareness.

Optimization of production structure type NDI demonstration is also a consideration . Obviously, cost would prevent numerous test specimens, therefore flexibility is reduced as to flaw location configurations. The test specimens would have to contain at least one flaw of each flaw size range being evaluated.

As an example, fifteen complex concept evaluation specimens similar to Figure 75 (comprised of 2 spars, 2 bulkheads, and 1 skin) containing at least 6 flaws, each of which is in a different flaw size range, can be utilized by running each specimen twice through 5 inspectors. This would result in:

5 inspectors x 15 specimens x 2 times x 6 flaws per specimen =
900 trials per NDI evaluation

or, 150 trials per flaw size.

The 15 test assemblies would contain open holes and holes with fasteners. Some of the six flaws would be bolt hole flaws.

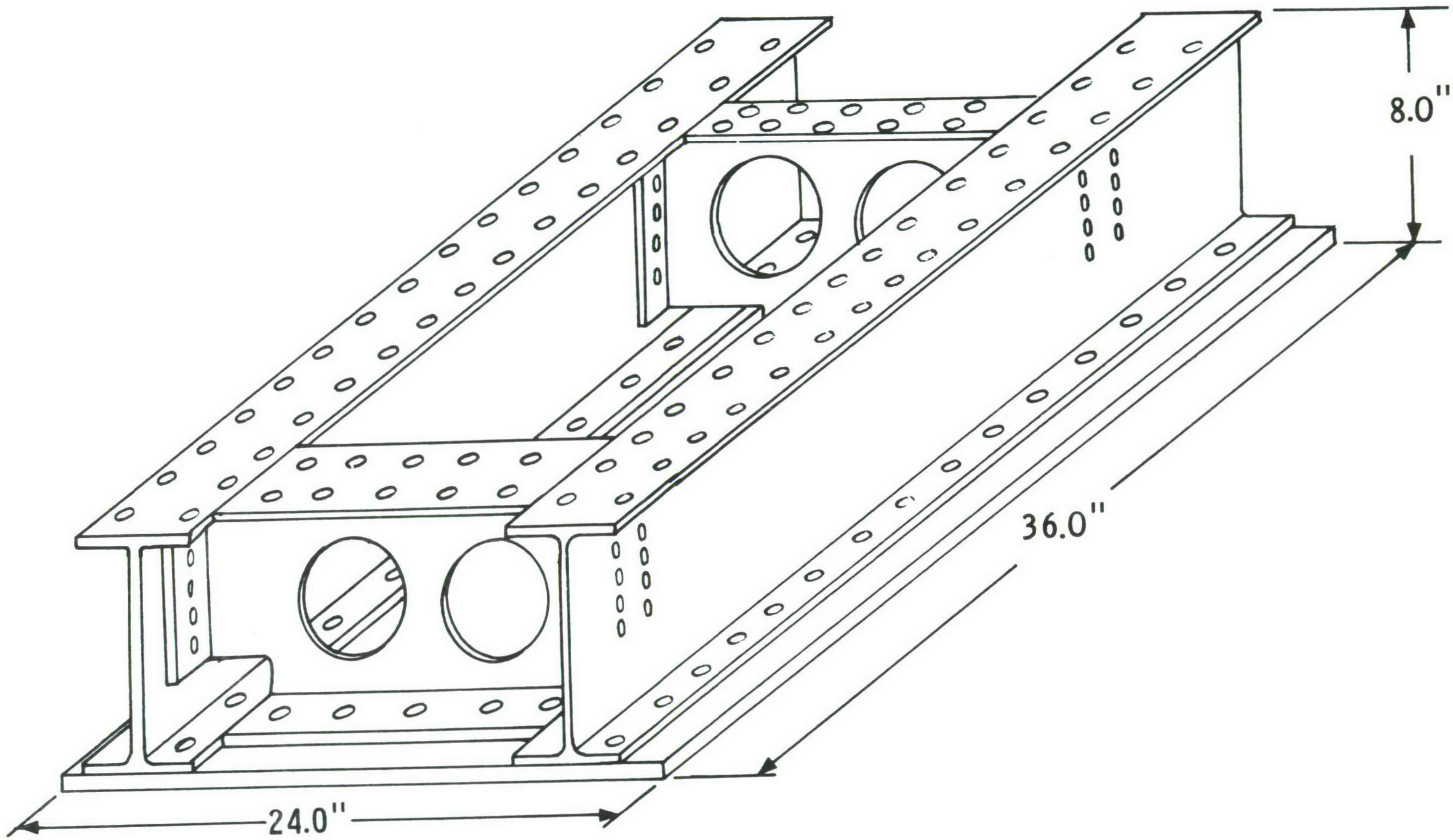


Figure 75 Complex Concept

The flaws would require creation in the details prior to assembly, then a close inspection after assembly to assure no new flaws or changes in the original flaws were introduced. Both rivets and bolts can be used as fasteners.

The simulated production part approach may eliminate the need for flaw-free control specimens because of the increased surface areas, fasteners, joints, and general configuration that would be flaw free.

The resultant complexity of manufacturing assembly inspection, and test evaluation of the complex specimen would cost much more than that of a simpler specimen such as the "T" specimen shown in Figure 76 (specimen described in paragraph IX.3.7.2.1). The relative cost increases are summarized as follows:

Program Tasks	Cost Increase Factor
Fabrication	12 to 15
Crack Inducement	6
Test Preparation	2
Demonstration Inspections	4
Data Analysis	1
Materials	0.5

IX.3.7.1.3 Difficulty of Inducing Cracks in Test Specimens of Complex Geometry. The two primary problem areas in inducing cracks are loading of the complex test specimen and control of the flaw size. Acquiring correct bending and load cycling parameters to propagate the crack from an induced stress concentration (or EDM slot) under controlled conditions is a very difficult achievement. This is very true where webs and stiffeners with cutouts are involved which would be typical for a complex test specimen. A high degree of scrap generation is to be expected. On the successful test specimens, restoration to the design criteria after fatigue cycling would still remain to be accomplished. Straightening and finish machining operations are required which again could generate further scrap.

IX.3.7.2 Proposed Demonstration Program

Based on the discussion above, a demonstration program has been developed for 2024-T851 aluminum structure in the baseline F-111F wing box which avoids some of the problems of a complex specimen program. It is specifically designed to verify NDI detection capability at the detail part and assembly levels of airframe manufacturing. The demonstration will be accomplished utilizing specially fabricated test specimens containing fastener holes of sizes and types representative of production article installations. Fatigue-type cracks of desired lengths will be randomly located around the circumference of the specimen bolt holes and in the surface of the specimens. The test specimens will be submitted to inspection personnel under manufacturing conditions utilizing typical manufacturing equipment and procedures. Results of these evaluations will provide the data necessary for a quantitative measure of manufacturing nondestructive inspection detection capability.

The demonstration will utilize a minimum number of simple geometry specimens. Costs of the program are discussed in paragraph IX.3.8 as part of the Fracture Control Plan assessment. For more complex specimens the program costs can be estimated using the factors discussed in paragraph IX.3.7.1.2.

IX.3.7.2.1 Detail Program Plan.

Test Specimens Design and Manufacture

The test specimens to be utilized for the demonstration will be fabricated from 2024-T851 Aluminum. Test specimen configurations (flat plate and "T") are shown in Figure 76. The bolt holes will be of one diameter (.312). Thirty-two (32) test specimens will be utilized with at least 25 of the specimens containing one flaw induced randomly around the circumference of a bolt hole. The 64 test specimens will be designed into 6 configuration groups combining two or more flaw conditions (reference Table XVII). At least 31 specimens of each group shall have induced flaws (i.e., one specimen has no flaws). All machined surface finishes will be 125 RMS. Hole finishes will be 64 RMS.

The induced cracks in the flat plate and "T" specimen surfaces (including "T" specimen radii) will be initiated by introducing EDM to local areas. Then the area will be subjected to cyclic loading until the desired flaw length is obtained. It is assumed that a semicircular shaped flaw will

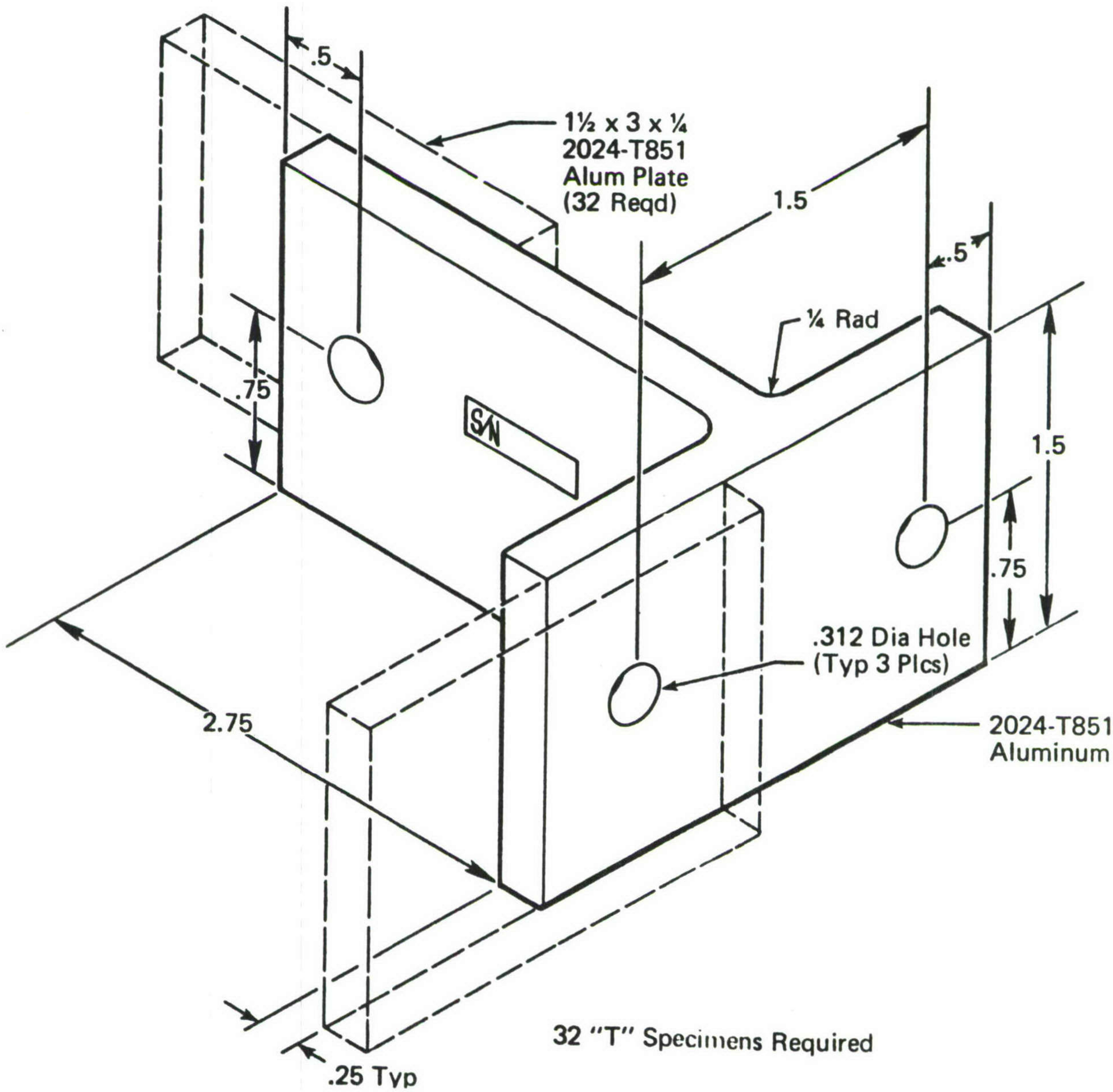


Figure 76 Specification Configuration

Table XVII
CONFIGURATION/FLAW SIZE PLAN

Specimen Configurations*	Flaw Surface Length
1. Flat Plate	
Type I	.010 to .030
Type II	.010 to .030
2. "T"	
Type I	.010 to .030
Type II	.010 to .030
Type III	.051 to .250
3. Two Flat Plates bolted	
Type IV	.031 to .050
Type V	.051 to .250
4. Two Flat Plates bolted	
Type VI	.031 to .050
Type III	.051 to .250
5. Flat Plate bolted to "T"	
Type III	.051 to .250
Type IV	.031 to .050
Type VII	.010 to .030
6. Flat Plate bolted to "T"	
Type V	.051 to .250
Type VI	.031 to .050
Type VII	.010 to .030

SUMMARY

6 Configurations @ minimum of 31 trials per configuration is a minimum of 186 trials. It is planned to conduct 150 trials per configuration for a total of 900 trials.

* Specimen types are defined in Figures 77 and 78.

be generated. The cracks in the holes will be initiated by EDM in a pilot size hole then finish reamed after sufficient growth is achieved through cyclic loading. All specimens will be finish machined, after crack achievement, to final dimensions as shown in Figure 76. No single test specimen shall contain more than two induced flaws, thus with a "T" specimen bolted to a flat plate, a maximum of 4 flaws is obtained. Using this approach, bolted test specimens can contain 0, 1, 2, 3, or 4 flaws.

Flaw Verification

Both prior to the demonstration and subsequent to the demonstration, the test specimens shall be cycled through the NDI Lab for flaw verification and mapping purposes. During these evaluations each flaw in the test specimens shall be subjected to a variety of nondestructive tests and measurements to confirm flaw length and location. At this time, a flaw location map will be prepared for each specimen depicting these determinations.

Review of Inspection Procedures

To assure that representative manufacturing evaluation results will be obtained, existing specific and detailed NDI procedures will be reviewed. This action is to assure adequate instructions for the inspectors involved during the NDI detection capability demonstration. Manufacturing operation planning sheets will be prepared in the same format as used for normal production operations and will provide routine step-by-step instructions of how to prepare the specimens for NDI and how to perform the NDI of the test specimens.

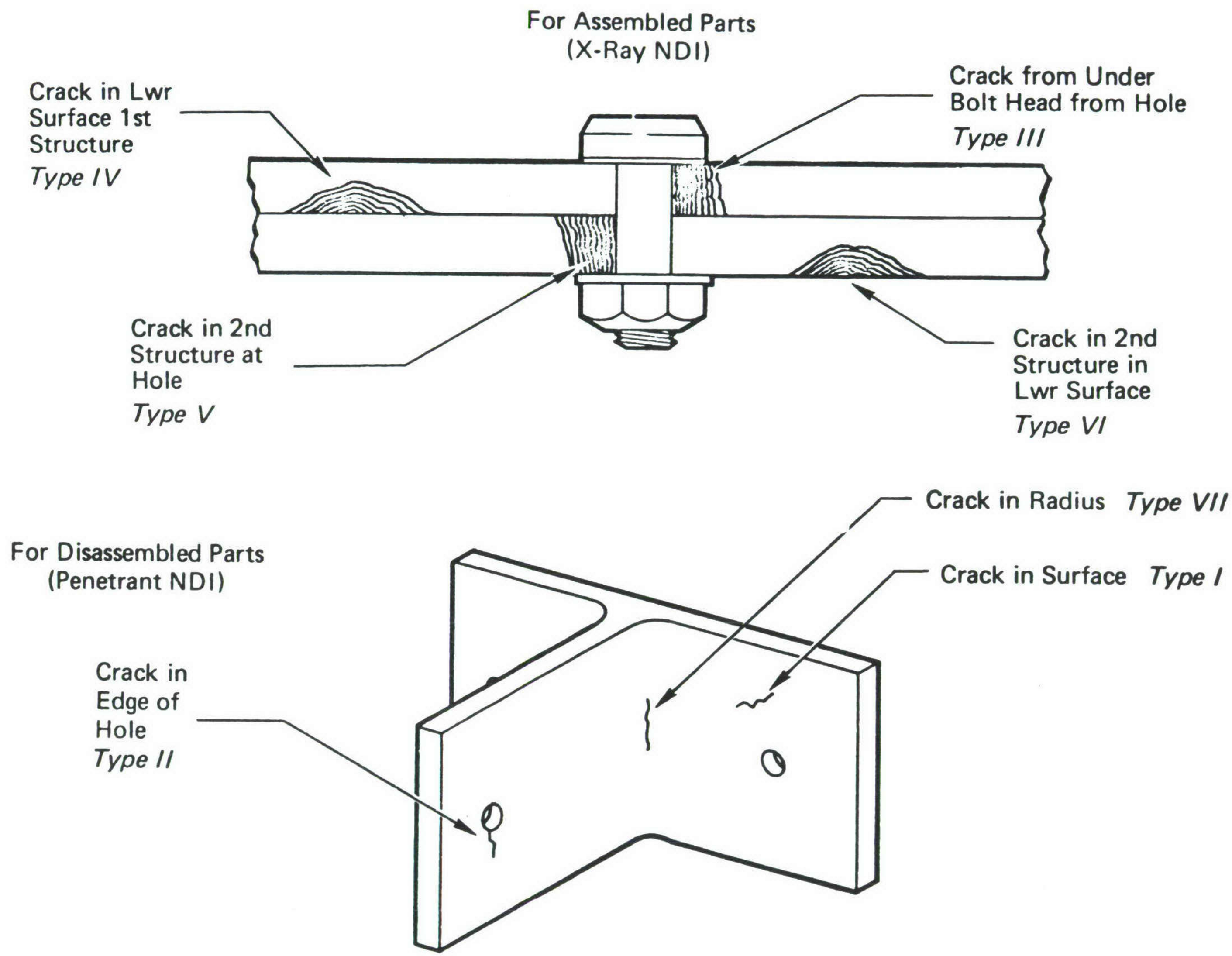


Figure 77 Trial Setup Types

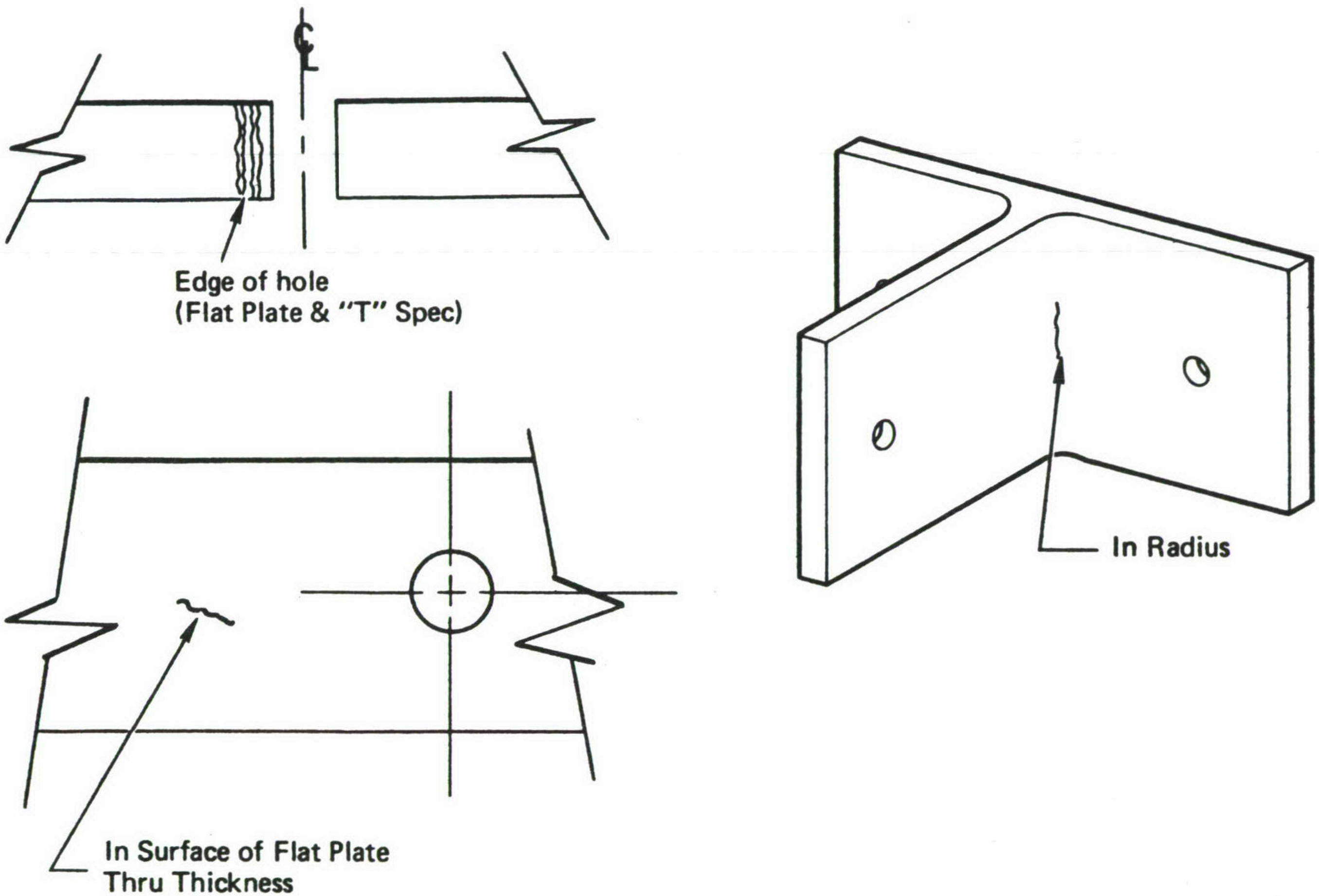


Figure 78 Crack Configurations

Demonstration Procedure

The basic steps to be followed in conducting the actual manufacturing capability demonstration are as follows:

1. The evaluator will select a predesigned set of test specimens with paperwork and will present them to Manufacturing Control.

   (Example: Thirty two specimens of configuration V).

2. The test set will be presented by Manufacturing Control to the inspector(s) certified to perform the NDI technique being evaluated. Manufacturing Control will also present the copies of the applicable procedures, planning and data forms.

3. The inspector(s) will set up the NDI equipment and perform inspection of the test specimens in accordance with the procedures and planning provided.

4. The inspector(s) will complete the data forms depicting the location and response of rejectable flaw indications observed during the inspection.

5. Manufacturing Control will pick up the test specimens and completed data forms and return them to the evaluator.

6. A comparison of the completed data forms with the master flaw location map for each specimen evaluated will reveal which of the induced flaws were detected and recorded by the inspector(s) and which were undetected.

7. Results of these evaluations will not be divulged to the inspector(s) involved.

Accumulation and Analysis of Data

A master log will be maintained by test specimen number for each specimen which will reflect the results of all evaluations. Specifically this log will reflect the cumulative number of trials and detections for each flaw in the test specimens. For purposes of this demonstration a trial is defined as the presence of one induced flaw available for

detection by one inspector. For example, if a sample set of 32 test specimens, 31 of which each contained two induced flaws, was presented to an inspector and he detected 60 flaws, this would constitute 62 trials and 2 misses.

Utilizing the 64 test specimens and approximate 4 to 5 inspectors, a sufficient number of evaluations will be conducted to yield a valid measure of the detection capability of the NDI technique for fatigue cracks. The demonstrated 95% confidence level minimum probability of detection will be calculated by use of the CHI-square (χ^2) distribution approximation of the binomial distribution. This technique provides the upper confidence limit for the mean number of failures (misses) with a required confidence (C X 100%) as follows:

$$nq = 1/2 \ \chi_c^2 \ , \ f = 2 \ (X_o + 1)$$

Where:

χ_c^2 = C - fractile of the χ^2 distribution

f = degrees of freedom for χ_c^2

X_o = number of failures (misses)

n = total number of tests (trials)

q = upper confidence limit for the probability of failure

1-q = lower confidence limit for the probability of success (p)

C = confidence coefficient

This mathematical approach is depicted graphically on the following page showing the minimum probability of detection at a 95% confidence level as a function of the number of trials and number of trials and number of detections.

As shown on the preceding graph (Figure 79) a minimum of 31 trials with 0 undetections are required to demonstrate a 0.9 minimum probability of detection for a given flaw length. For purposes of this program wherein inspectors operating in a manufacturing environment will be evaluated, it is desired to accumulate 150 trials for each configuration

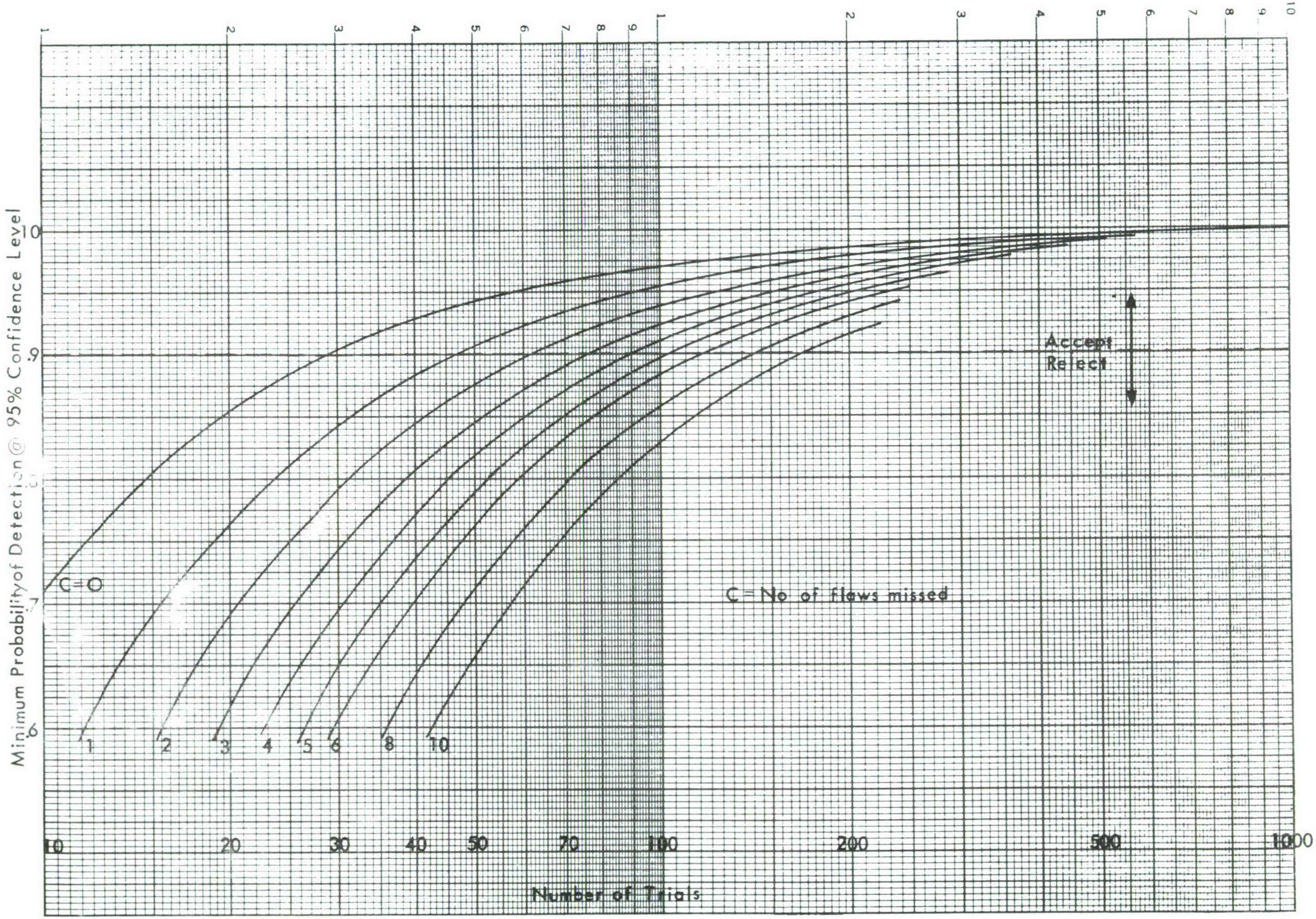


Figure 79 Flaw Detection Probability

combination evaluated to compensate the human factors elements which will affect the evaluation results. This value (150) is based on data obtained from the Human Factors Program conducted by Convair Aerospace, Fort Worth Operation per F-111 ECP 10544, Nondestructive Inspection Improvement program. If no more than 8 undetections are experienced during the accumulation of each of these 150 trials, the minimum probability of detection will be demonstrated to be equal to or greater than 0.9 for each of the cases.

Normalization of Inspector Attitudes

Any inspector, from routine, will be familiar with what parts or assemblies that flow through his area. Any deviation will create a certain amount of apprehension. However, the inspectors are accustomed to an occasional flow of engineering test parts which are used for design verification tests. This approach will be utilized for the NDI detection demonstration. Engineering drawings shall be prepared and released through routine channels. Manufacturing work instructions identical to production parts shall be prepared along with production NDI technique data sheets. The Quality Assurance NDI Specialist who normally services the area during production work shall be utilized to monitor the demonstration (i.e., no strangers present).

Destructive Analysis of Test Specimens

Upon conclusion of accumulation of the desired number of evaluation trials, a representative number of the specimens will be sectioned in order to precisely measure and photomicrograph the induced flaws in order to verify the evaluation and original mapping data.

Preparation of Demonstration Report

Upon conclusion of all tests and analyses a complete formal report will be prepared detailing all phases of the program, including detail evaluation results, induced flaw photomicrographs and data, data analysis, and program results and conclusions. A draft of this report will be submitted to appropriate customer personnel within 6 weeks of program conclusion. Within 2 weeks of receipt of draft approval, the final report will be published.

If the 90% probability of detection at a 95% confidence level criteria is not achieved for the smaller flaw lengths, the actual achieved probability of detection level and confidence level will be reported as the NDI detection capability.

IX.3.8 Impact of Fracture Control Program

The purpose of this section (reference Paragraph 3.1.1.1.7 in FZP-1402 Addendum 1) is to review and analyze all aspects of the fracture control program given in MIL-STD-1530 and MIL-A-8866A as they would apply to the baseline. In addition, the cost of applying the fracture control program to the baseline wing will be estimated.

To fulfill these requirements, a detailed list of elements for a fracture control plan applicable to the F-111 baseline wing was prepared to reflect new requirements. This list is discussed in paragraph IX.3.8.1 below. A copy of MIL-STD-1530, dated 1 September 1972, is given in Section IX.7.

IX.3.8.1 Fracture Control Plan Tasks

The following paragraphs define those tasks associated with a fracture control plan as it would apply to the baseline wing box. A detailed breakdown of work by each major discipline is described and costed.

IX.3.8.1.1 Damage Tolerance Design Concept/ Material/Cost Trade Studies. Trade studies depicting weight and cost as a function of design concept and material selection will be conducted during preliminary design to obtain cost and weight efficient designs which meet the damage tolerance requirements.

Design selection will be based on meeting all the integrity and reliability requirements and the results of the trade studies. Reference paragraphs 3.1.1.1.1, 3.1.1.1.2, 3.1.1.1.3, 3.1.1.1.8.2 and 3.1.1.1.9 of FZP-1402 Addendum 1 for a definition of additional tasks associated with trade studies.

The fatigue and fracture analysis group will provide design allowable curves for flaw types specified by the criteria:

- Part through cracks (3 thicknesses)
- Through cracks (thin sheet)
- Bolt hole thru cracks (2 diameters)
- Bolt hole corner cracks (2 diameters)

Allowable curves will be prepared for each of three

different material candidates. Six (6) stress level variations will be needed to define each allowable curve. Using a flight-by-flight spectrum, six stress levels, eight flaw types, and three materials will result in 144 growth curves.

The structural analysis group will utilize the trade study results (allowable stress curves) and determine weight variations in the baseline wing structure. Weight variations with design allowable stress in the lower surface will be plotted.

Fatigue analysis design allowables curves will be prepared to aid preliminary design. Baseline preliminary fatigue analysis are normally performed on the final configuration, but design allowable curves are a direct result of the new criteria. Eight fatigue damage curves (factors on stress level) for four K_t values will be prepared using Miner's damage rule.

IX.3.8.1.2 Basic Fracture Data. All materials property data used to qualify fracture critical parts in accordance with the Fracture Mechanics Design Requirements will be documented as a test plan attached to the Fracture Control Plan. These data will include mechanical properties, fracture toughness, cyclic crack growth rates, and stress corrosion cracking thresholds, as applicable. Also included will be spectrum/environmental tests required to establish spectrum (FLT X FLT) interaction (retardation) effects. Surface flaw and bolt hole flaw tests will be included. All test data generated in the test program will be compiled and included in the test plan as a separate addendum.

Available materials data on candidate materials will be collected and documented. A test program will be designed, executed, and documented to obtain needed fracture data including K_{Ic}, K_{Iscc}, da/dN and spectrum/environment effects.

Spectrum environmental tests to aid in developing new retardation model or in determining "m" exponent for the Wheeler model will be conducted. Approximately 15 surface flaw and 15 bolt hole flaw specimens will be tested using a flight-by-flight spectrum. Results will be analyzed by computer.

During the manufacture of parts listed as fracture critical, test coupons must be removed and tested from each part. The costs for this requirement would be classified recurring costs and would be spread over the life of the production program.

IX.3.8.1.3 Fracture Critical Parts. A part is defined as fracture critical if catastrophic failure of the part would result in loss of the aircraft.

Critical parts shall be selected by a review of primary structure which is principally loaded in tension and experiences exposure to a corrosive environment.

The review shall be a joint effort by the Structural Design and Analysis Groups and the Fatigue and Fracture Analysis Group. The review shall result in a fracture critical parts lists which shall be updated on a systematic basis as the design evolves. Trade study results, including initial damage sizes, will be reflected as they become available and revisions made as necessary to the parts list.

The critical parts list shall provide the following information for each part as a minimum:

1. Part description and location in the structure

2. Drawing number

3. Type of material and basic form

4. Type of fabrication applied to the part, if any.

The critical parts list shall be maintained and updated as required by the Structural Design and Analysis Groups. The list shall be distributed to supporting groups and reissued as revisions are made.

IX.3.8.1.4 Design Drawings for Fracture Control Parts. The engineering drawing is the single means of transmitting the requirements of the fracture control plan. Fracture critical parts will be identified by the following drawing note:

> This part is categorized as a fracture critical part and is subject to all requirements of the fracture control plan.

Material procurement and material processing specifications along with NDI and corrosion protection requirements will also be specified on the drawings for all fracture critical parts. No deviation from these drawing requirements will be permitted without a corresponding change to the fracture control plan.

Drawings in which only portions of the part is categorized as "fracture critical" will be zoned to identify these areas. These areas will refer to a note on the face of the drawing which will read:

This zone of the part is categorized as a fracture critical zone and is subject to all requirements of the fracture control plan.

Fracture critical parts processed in accordance with toughness controlled specifications will include a test tab for verification of fracture toughness subsequent to processing. The drawing will identify and locate the test tab.

Typical drawing notes for a fracture critical part are as follows:

1. This part (zone) is categorized as a fracture critical part (zone) and is subject to all requirements of the fracture control plan.

2. Serialized traceability is required.

3. The material must meet the special requirements of ______________________________.

4. Braze (bond, weld, etc.) per ____________________.

5. Special corrosion protection required per ______________________________.

6. Perform NDI in accordance with ____________________.

7. Fasteners shall be installed and inspected in accordance with ____________________.

During preliminary and production design, Quality Assurance will review drawings to insure that all inspection and maintenance requirements are documented.

IX.3.8.1.5 Establish Complete NDI Requirements, Process Control Requirements, and Corrosion Requirements for all Fracture Critical Parts. The quality assurance system applied to components on the Fracture Critical Parts List shall insure that materials and parts conform to the engineering drawing. Inspection and process control requirements for the materials, detail parts, and assembly will be defined as the detail designs and manufacturing plans are developed. These requirements will be incorporated into a Quality Assurance (Q/A) Plan for Production.

Inspection - A comprehensive inspection plan will be followed to provide a high degree of confidence that no defects are present in the raw material or introduced during fabrication which would cause premature failure of the part. Engineering will support Quality Assurance with NDI instructions for each critical part; with design of pre-flawed test specimens (shape and orientation of flaws); and with interpretation of damage tolerance criteria requirements.

During production, engineering support for consideration and disposition of discrepant fracture critical parts and flaws in fracture critical areas of assemblies throughout production.

Receiving Inspection will verify that the raw material meets the requirements of the material procurement specifications applicable to fracture critical parts. This will be accomplished by reviewing the records supplied by the vendor and by conducting the applicable acceptance tests. Traceability records will be initiated by transferring the heat numbers (for steel and titanium) or lot numbers (for aluminum) to the receiving report (RR, FW 159).
(RR, FW 159).

Material will be marked nonconforming if the supplier has not complied with the quality, documentation or traceability information requirements or if the material fails to meet the acceptance test requirements. Disposition of nonconforming material will be made by the Material Review Board.

Detail part inspection will be in accordance with specific NDI instructions prepared for each fracture critical part prior to production. To assist

in preparation of these instructions, preflawed test specimens simulating each of the fracture critical parts will be fabricated and subjected to thorough inspection. The NDI method selection and procedure development will be based on the ability to detect these known flaws. Inspection personnel will be trained and qualified for any new inspection techniques before inspection of the product is required.

Implementation of the detail part inspection procedures will be under the direction of Quality Assurance engineers. Difficulties with the inspection instructions, the NDI equipment or procedures, etc., encountered during production will be resolved by implementation of the necessary corrective measures. Product discrepancies will be documented on Quality Assurance Rejection Reports along with failure diagnosis, disposition as determined by the authorized material review authority, and action taken to correct the cause of the discrepancy.

Final assembly inspection provides verification of the material, dimensional and installation requirements specified on the engineering drawing. In addition specific inspection instructions will provide for the in-process controls necessary for assuring that quality requirements are met. Manufacturing operations such as drilling, torquing, shimming, etc., will be monitored. All holes drilled in fracture critical parts will be inspected in accordance with surface finish, roundness, edge condition and protrusion (where applicable) requirements established by Engineering Standards. Torque requirements and shim allowables will be specified on the Engineering drawing. Tools and manufacturing aids will be inspected to the extent necessary to minimize rework and to eliminate undue installation stresses. Corrosion control requirements for sealing, cleaning and finishing will be verified.

<u>Process Control</u> - Production processing and fabrication will be controlled by documented work instructions, adequate production equipment and calibrated monitoring devices. Special testing requirements invoked by toughness controlled processing specifications will be accomplished.

Documented work instructions that conform to engineering Process Specifications revised to incorporate the new criteria will be used for all production, processing and fabrication work. Adherence to these instructions will be certified by inspectors, and coupled with satisfactory product inspection, will serve as verification of acceptable

workmanship. Production procedures in current use are described by Process Standards.

Processing variables such as temperature, time, atmosphere, solution composition, etc., will be monitored using calibrated devices. For multiple-step processes such as brazing, bonding and welding, Quality Assurance Data Sheets will be prepared to document the concurrance of the processing variables for each step of the process. Process Standards will define acceptable ranges of temperature variation in heat treat furnaces, chemical composition in solution tanks etc.

Corrosion Control - The corrosion protection and control plan developed for the F-111 program will be used. Elements of this plan include (1) judicious selection of the basic materials of construction (2) selection of processing variables and heat treat conditions that minimize stress corrosion and hydrogen embrittlement; (3) application of antifretting coatings to all faying surfaces; (4) special precautions in areas involving dissimilar metal contact; (5) choice of the finish system best suited for the individual design; and in summary (6) careful attention to design detail wherever the possibility of corrosion exists.

The basic documents for this plan are the Finish Specification, the Finishing Code, the Sealing Specification, and the Design Standards for Integral Fuel Tanks. The Finish Specification covers the procedures to be followed for protecting from corrosion. The methods and materials required for cleaning, surface treatment and application of finishes and protective coatings to parts are described, and applicable military and General Dynamics specifications are referenced. The Finishing Code is the engineering drawing call-out procedure that completely defines the finishing requirements for the production planners. The Sealing Specification specifies materials, equipment, and procedures for sealing integral fuel tanks and shall be specified on all applicable drawings which involve fuel tank sealing.

Standard notes, drawing call-outs and List of Material entries are contained in the Drafting Room Manual. The Design Standards for Integral Fuel Tanks contains design information such as a description of the multiple barrier sealing system that is used.

The existing corrosion control plan will be supplemented by drawing notes to cover special cases that arise because of new design concepts.

IX.3.8.1.6 NDI Demonstration Program. Potentially, an NDI demonstration program can reduce the size of the assumed initial damage. This translates directly into lower weight and/or increased service life as discussed in paragraph IX.3.7 above. Costs for NDI demonstration program are directly affected by the type of specimen chosen. Generally there are three categories of specimens which can be considered:

(a) A simple geometry specimen program (Flat Plate)

(b) A moderately complex part such as "Tee's" or "H" section specimens

(c) Use of actual structural components with induced flaws

The demonstration program defined in paragraph IX.3.7.2 is based on the type (b) specimens. For type (c) specimens the program is considerably more expensive as discussed in paragraph IX.3.7.1.

IX.3.8.1.7 Material Procurement and Manufacturing Process Specifications. Material procurement and processing will be controlled by a series of specifications which are sufficient to preclude use of materials which have properties inferior to those assumed in design.

<u>Fracture Toughness Material Acceptance Criteria</u> - Materials used in fracture critical parts will be purchased to a guaranteed minimum fracture toughness requirement. A plane strain fracture toughness test will be conducted in accordance with ASTM specification E399-70T for all material purchased in thickness greater than the ratio $\left(\frac{K_{Ic}}{F_{ty}}\right)^2$ where K_{Ic} and F_{ty} are minimum guaranteed values for plane strain fracture toughness and 0.2 per cent offset yield strength, respectively. For thinner gages, there are no specific fracture toughness requirements; however, the

processing procedures and the controls on chemistry and microstructure used to virtually assure adequate toughness will be specified for all thickness levels.

The ASTM specification for valid K_{Ic} measurements requires that the specimen thickness exceeds $2.5\left(\frac{K_{Ic}}{F_{ty}}\right)^2$

Test results where $\left(\frac{K_{Ic}}{F_{ty}}\right)^2 < t < 2.5\left(\frac{K_{Ic}}{F_{ty}}\right)^2$ will be reported as K_Q values. Acceptable K_Q values may provide reasonable assurance that the material has properties in excess of those assumed in design. Use of the ASTM E399-70T procedures for specimens thinner than $\left(\frac{K_{Ic}}{F_{ty}}\right)^2$ is not recommended because of the possibility of computing reduced apparent toughness values caused by unstable growth of the plastic zone.

<u>Material Procurement Specifications</u> - Material procurement specifications will be prepared for each material and product form planned for usage in fracture critical parts. These documents will specify minimum acceptable values for the tensile properties and the fracture toughness. Requirements will be established for melting and primary processing, heat treatment, chemical composition, ultrasonic quality and dimensional tolerances. Quality assurance provisions, testing procedures and reporting requirements will be described. A microstructural requirement will be established for materials where the structural property relations are sufficiently understood.

<u>Processing Specifications</u> - Processing specifications will be prepared for all material process combinations planned for usage in fracture critical parts. As a minimum, specifications will be prepared for heat treatment, adhesive bonding, brazing and welding. Requirements will be established for equipment control, processing procedures, acceptance standards, quality assurance provisions and workmanship. Where applicable, mechanical properties, chemical compositions, surface requirements, and dimensional tolerances will be defined. Procedures for items such as surface preparation, rework and repair, and temperature control will be defined in detail. Applicable testing procedures, subcontractor

provisions and reporting requirements will be covered.

IX.3.8.1.8 Materials Traceability. For each fracture critical part, complete data documenting the raw material heat number, manufacturing planning, inspection records, discrepancy reports will be recorded, collected and maintained, as described in Standard M186 (reference Section IX.7). These records will provide complete traceability of produce quality from raw material through the completed assembly. Traceability will be implemented in accordance with Standard Practice 9-23.1.

In order to trace raw material through all processing, the vendor heat or lot number is related to the first shop order serial number and part number at first cut level. The shop order serial number identifies the part through to the end item.

IX.3.8.1.9 Requirements for Damage Tolerance Analysis and Testing Activities as Specified in MIL-STD-1530, MIL-A-8866 and MIL-A-8867 must be met. Conformance to the structural design requirements requires accurate and timely structural analysis. Stress analysis is essential for the determination of the static strength, fatigue life and damage tolerance. Fracture analysis will be conducted in accordance with the procedures outline herein.

<u>Stress Analysis</u> - Finite element procedures will be the principal tool used in the stress analysis work. Several programs having a wide variety of elements are available for use. The output data from the finite element analyses will be obtained in both printed and plotted form. The accuracy of the input data will be validated by plotting the geometry data and by checking the input loads data against internal loads obtained from finite element stress distribution.

Conventional methods of analysis using manual or programmable calculators will be used as required to supplement the finite element work. For example, distribution of the applied loads to the node points of the finite element models will be done by manual methods.

Fracture critical locations will be identified on the basis of the coarse grid finite element analysis. Fine grid stress analysis will be conducted in these locations to determine localized stress concentrations and to better define specific control points for fracture analysis.

Fail-Safe Analysis - The same finite-element math-models used for stress analysis will be used to conduct a residual strength analysis of the five spar wing fail-safe design of the baseline. In the complete finite element simulations, individual elements can be reduced in size or eliminated to simulate failure. Orthotropic elements are used to simulate inability to react shear and normal forces along a crack or line of separation. Stress distributions in the altered structure are plotted and tabulated using the stress-analysis output format.

Crack Growth Analysis - A crack growth analysis will be conducted to determine the safe crack growth characteristics. An initial crack of the size specified in the criteria is assumed to exist in the most unfavorable orientation with respect to the applied stress and the material properties. The growth of this flaw in the anticipated chemical, thermal and cyclic-stress environment will be computed using constant amplitude crack growth data and an analysis model that satisfactorily accounts for load interaction effects due to variable amplitude fatigue cycling.

Control points will be defined for the primary tensile-loaded elements in the baseline wing box. These control points will be selected on the basis of the finite element stress analysis results and a consideration of the design detail. For each control point, the functional relationship between K and crack length will be defined using existing stress intensity models coupled with estimating techniques. Experimental data that relate the crack growth rate and critical crack size to the applied stress intensity level shall be generated as discussed in the Material Test Plan. Crack growth will be calculated using the baseline service loads fatigue spectrum by integrating the growth rate between the limits set by the assumed initial flaw size and the critical flaw size. The Wheeler crack growth model will be used to account for load sequencing effects.

The value of the retardation exponent, m, required for the Wheeler model will be determined empirically in the spectrum fatigue test program defined in the material test plan. Thermal effects on crack growth rates will be neglected; however, the critical crack size will be based on the fracture toughness at the minimum structural temperature established for the baseline. Sustained load crack growth, da/dt, will be assumed negligible providing stress intensity computed using the lg stress level and the critical crack length is less than the stress corrosion threshold, K_{Iscc}.

The allowable spectrum stress for control points in monolithic parts will be determined on the basis of crack growth analysis. Curves will be prepared by plotting the maximum allowable spectrum stress as a function of the assumed initial flaw size and the specified inspection interval. The specific methodology for generating stress allowables curves is as follows:

1. Calculate a series of crack growth curves (crack length vs number of flights) using a series of factors on stress level.

2. From (1) determine the maximum initial crack size that permits one inspection interval of subsequent growth as a function of the maximum stress in the spectrum.

3. Plot the allowable spectrum stress as a function of initial flaw size for inspection intervals of 1/2, 1 and 2 lives.

4. Determine the allowable spectrum stress level in accordance with the initial flaw size and inspection interval requirements of the criteria.

Critical crack size will be calculated for each control point using the stress intensity functions defined for the crack growth analysis.

Plain strain fracture toughness, K_{Ic}, will be used for critical crack length calculations in control points classified as plane strain and mixed mode. Plane stress fracture toughness, K_c, will be used for critical crack calculations for control points classified as being in a plane stress state.

A preliminary and a final analysis will be prepared on the final design configuration. Each part on the critical parts list will be classified according to damage tolerance approach and degree of inspectability. A crack growth analysis necessary to demonstrate preliminary compliance of the final design with the requirements selected as applicable to each critical part.

A series of fifteen analyses will be performed for bolt hole flaws (one diameter), surface flaws (two thicknesses) in spar caps and lower skins, a pylon cutout, and a fuel flow hole configuration.

In addition to preliminary analysis above, a final crack growth analysis of the completed wing is required to calculate final inspection intervals. Full scale test results will be reflected in this work. Approximately 5 control points will be used for this analysis.

Structural Testing - A comprehensive development test program is required in conjunction with the design and analysis tasks. Development test in support of damage tolerance, and NDI evaluation are required using a flight by flight spectrum. In addition, proof of compliance damage tolerance testing is required on the full scale fatigue test article using a flight-by-flight test spectrum. Damage tolerance testing will be conducted following completion of the fatigue test program. Emphasis will be placed on the following test categories:

A. Preliminary Design - Concept Verification Testing

During preliminary design, additional testing will be required to establish basic design concepts, material selection, and configurations. These are listed below:

- Static testing - Subscale element tests to verify residual strength

- Damage tolerance testing - Subscale element tests to show crack growth behavior under flight-by-flight spectrum loading

Test results, including da/dn measurements, will be analyzed and interpreted and used to adjust preliminary analysis of structure.

B. Detail design - Pre-Production Validation Testing

Further testing to satisfy new damage tolerance criteria is described below:

- Static testing - Box beam type specimens for residual strength determination.

- Damage tolerance testing - Box beam type component specimen for verification of flaw propagation.

Analysis is required to define flaw location, type, shape, and orientation. Strain gage locations will be specified on the test drawings. Damage tolerance testing will utilize a flight-by-flight spectrum.

C. <u>Full scale test articles</u>

Full scale proof of compliance testing will include fail safe residual strength tests (performed on the static test article) and damage tolerance (flaw propagation) testing performed on the fatigue test article. Damage tolerance testing will be conducted using a flight by flight spectrum.

IX.3.8.1.10 Procedures as Required for Depot Level Inspection or Special Field Service Inspections Including (1) Inspection, (2) Maintenance and (3) Testing Will Be Developed for Fracture Critical Parts. The damage tolerance requirements in the proposed criteria are a function of inspectability. Inspection requirements are also given in the criteria for each degree of inspectability. Inspection requirements given for depot level for monolithic (slow crack growth) structure such as the baseline are further defined as to whether inspection must be performed on the part while installed or removed.

In service inspections must be documented in appropriate handbooks. NDI procedures must be developed, and written instructions prepared for field level and depot level inspection of critical parts.

IX.3.8.1.11 Chemical and Thermal Environment Definition. The definition of the baseline chemical and thermal environment for use in fracture analyses is covered by paragraph 3.1.1.1.4 of FZP-1402, Addendum 1 to the ADP Wing Contract. However, the estimated costs of accomplishing environmental definition for the baseline will be provided to supplement the data in 3.1.1.1.4.

The transient temperature distribution within the wing box structure will be calculated at approximately ten different locations.

IX.3.8.1.12 Risk Assessment Analysis. The evaluation of structural in-flight risk is accomplished using statistical and probability techniques. Specifically the risk assessment analysis can be used to:

- evaluate individual aircraft or fleet structural probability of survival during service life considering initial NDI/proof test prior to operational usage and considering subsequent inspections during service life

- establish which of the individual structural parts are most critical

- investigate the sensitivity of probability of survival values to variations in parameters such as usage, initial flaw size distribution, etc.

The major evaluation tool used in this analysis is the computerized risk assessment model. The risk assessment model is basically a set of mathematical and probability equations which describe a close approximation of the probability of structural survival during aircraft operations in the service environment. The equations are a function of those parameters that influence failure including:

- Initial flaw size distribution within each part which includes the influence of the non-destructive inspection (NDI) probability of flaw detection and of

proof test maximum flaw length.

- o In-flight flaw growth predictions which reflect service environment.

- o Time to failure distribution for a part with an initial flaw of given size which includes the dispersion of part times-to-failure due to variations in load history, crack growth, K_{Ic}, etc.

- o Periodic inspections accomplished at specified time intervals to insure the integrity of primary structure

The initial flaw size distribution, the flaw growth predictions, the failure distribution and the periodic inspection information serve as input data into the risk assessment model. The model translates these inputs to probability of survival values for a single aircraft and the fleet. These values are a measure of the inflight risk associated with the critical structure in the crack propagation failure mode.

In addition to these tasks, flaw sizes and crack growth curves for use in risk assessment of the final design must be provided. These curves will be supplied from the final crack growth analysis effort.

IX.3.9 In-Service Inspection

The object of this task is to collect in-service inspection information for the baseline wing structure and use this information in a parametric analysis to determine the impact on damage tolerance criteria.

IX.3.9.1 Baseline Wing - In-Service History

The Baseline Wing is the F-111F wing. Since the F-111F is the newest F-111 model in the AF inventory, only limited in-service information is available. For this study data was collected for all F-111 models for the time period of August 1969 through November 1972. Flight time for F-111 aircraft through this period was:

F-111A	69,735.7
F-111D	5,670.0
F-111E	13,427.9
F-111F	13,609.6
Total	102,443.2 Hours

Table XVIII shows various sources where F-111 in-service inspections could originate. The basic in-service data is obtained for TAC and SAC Base Maintenance Data Collection Records, AFTO Form 349 which is compiled into AFM 66-1 Maintenance Data. AFMM6-1 data is programmed monthly by Convair Aerospace into a computerized system known as Variable Inquiry System - Tape Oriented (VISTO). VISTO data for maintenance action on F-111 wings general, wing frames, wing skins and wing covers, for the period from August 1969 through November 1972 is shown in Table XIX. The necessary codes to read this computer data are shown as follows:

VISTO CODE	CODE TITLE	CODE LETTERS
Suffix	F-111 Model	FV = F-111A F5 = F-111D F6 = F-111E F7 = F-111F
WUC	Work Unit Code	Table XX
HOW MAL	How Malfunctioned	Table XXI
Action	Repair Action Taken	Table XXII
Man Hours	Time to Repair	
Units	Actual Number of Malfunctions of Type Shown for a given WUC code	

Each entry on the VISTO data which appeared to be a crack (How Mal Code 190) etc. for which the Action was 'repair' (Action Code F) was further checked by"dumping" the computer maintenance data for that item, In this data, part numbers are shown. In no case was there any maintenance action on the wing box structure during the time period August 1969 through November 1972.

A review of aircraft and engine operating limits (A & EOL) revealed no operating limitations on the wing box structure out of 921 A & EOL's for the F-111. A & EOL limitations are based on failure of test parts to meet certain criteria (ASIP requirements) and on general aircraft operating experience, i.e., problems encountered during use.

Table XVIII

IN-SERVICE INSPECTION F-111F WING BOX

SOURCE OF INSPECTION	FREQUENCY	NDI TECHNIQUE	ACCESSIBILITY	INSPECTION COVERAGE	ACTION TAKEN
1. TIME COMPLIANCE TECHNICAL ORDER (TCTO)	NO TCTO'S FOR WING BOX[1]				
2. TECHNICAL MANUALS:					
A. SCHEDULED INSPECTION AND MAINTENANCE REQUIREMENTS T.O. 1F-111F-6	PREFLIGHT[2]	VISUAL	LIMITED ACCESS TO BASIC STRUCTURE	WING SURFACE	INVESTIGATE ANY FINDINGS REPAIRS GENERALLY AT DEPOT LEVEL MAINTENANCE.
	"AFTER VIOLENT [2] MANEUVERS OR HARD LANDING"	VISUAL	LIMITED ACCESS TO BASIC STRUCTURE	WING SURFACE FOR CRACKS, SMOOTHNESS SHEARED OR PULLED FASTENERS, FUEL LEAKS	INVESTIGATE ANY FINDINGS REPAIRS GENERALLY AT DEPOT LEVEL MAINTENANCE.
B. STRUCTURAL REPAIR INSTR. T.O.1F-111F-3	SAME AS ABOVE[2]	VISUAL BOROSCOPE X-RAY	LIMITED TO FREE ACCESS DEPENDANT ON INVESTIGATION AND PARTS REMOVAL ACTION	WING SURFACE INVESTIGATION OF PROBLEMS MAY REQUIRE ADDITIONAL INSPECTION.	INVESTIGATE AND PERFORM STRUCTURAL REPAIR AS REQUIRED.
3. F-111 RECOVERY PROGRAM 12AEI-11-1047 (12AEI-11-1024 FOR F111A, E, D)	ONE TIME	VISUAL PRIOR TO AND AFTER COLD PROOF TEST[3]	LIMITED	WING SURFACE	
4. SECOND STRUCTURAL INTEGRITY PROGRAM (II SIP) FZM-13413	2000 HRS. FOR F-111F 1500 HRS. FOR F-111A, E, D)	VISUAL	LIMITED	WING SURFACE	
5. AIRCRAFT AND ENGINE OPERATING LIMITS (A & EOL)	NO AEOL FOR [4] WING BOX OUT OF 921 AEOL FOR F-111 AIRCRAFT				
6. IRAN PROGRAM (FUTURE)	NOT YET ESTABLISHED				

(1) TCTO 1F-111-627D 30 JUNE, 1969 CALLS FOR INSPECTION FOR IMPROPER SUB. OF TITANIUM BOLTS IN WING PIVOT SUPPORT STRUCTURE (MFG. PROBLEM)
(2) REFER TO SUMMARY OF AFM 66-1 DATA TO GAIN F-111 ACTUAL EXPERIENCE.
(3) X-RAY INSPECTION WAS USED OF F-111A, E, D, C, AND FB111 IN ORDER TO SHOW LOCATION OF MANUFACTURING INDUCED CRACKS IN SUBSTRUCTURE PARTS. THIS X-RAY INSPECTION IS A PART OF F-111F PRODUCTION.
(4) MANUFACTURING INDUCED CRACKS IN SUB-STRUCTURE PARTS CAUSED LIMITATIONS WHICH FELL WITHIN EXISTING LIMITATIONS ON A/C. ONCE REPAIRED, NO LIMITATIONS EXIST.

Table XIX

66-1 VARIABLE INQUIRY REPORT

AUG 1969 – MAY 1972

SUFFIX	WUC	ACTION	HOW MAL		MANHOURS	UNITS
FV F-111A	11BAA FRAME	G (R/R MINOR PART)	020 (WORN)	A	32.7	18
			070 (BROKEN)	A	4.0	1
			105 (DAM FAST)	A	197.2	62
			106 (MISS FAST)	A	60.1	13
			117 (DETERIOR)	A	137.0	1
			127 (ADJUST)	A	15.5	8
			129	A	2.0	1
			150 (CHATTER)	A	2.0	1
			190 (CRACKED)	A	110.2	10
			730 (LOOSE)	A	7.0	1
			780 (BENT)	A	4.0	1
			846 (DELAM)	A	40.5	2
			947 (TORN)	A	51.0	2
				B	663.2	121
		L (ADJUST)	020 (WORN)	A	.3	1
			106 (MISS FAST)	A	3.0	3
				B	3.3	4
		P (REMOVED)	190 (CRACKED)	A	.5	1
				B	.5	1
		Q (INSTL)	108 (WIRE/KEY)	A	.6	2
				B	.6	2
		R (R/R)	947 (TORN)	A	8.0	1
				B	8.0	1
		X (T-1-S)	190 (CRACKED)	A	7.5	1
			799 (NO DEFECT)	A	.6	1
				B	8.1	2
				C	683.7	131
	11BAB SKIN	F (REPAIR)	020 (WORN)	A	2.2	1
			190 (CRACKED)	A	35.0	3
			425 (NICKED)	A	12.0	1

Table XIX (Cont.)

65-1 VARIABLE INQUIRY REPORT

SUFFIX	WUC	ACTION	HOW MAL			MANHOURS	UNITS
FV	11BAB	F	780	12N7620/815			
F-111A	SKIN	(REPAIR)	780 (BENT)		A	16.0	
					B	65.2	5
		G (R/R MINOR PARTS)	020 (WORN)		A	9.7	5
			070 (BROKEN)		A	9.0	4
			105 (DAM FAST)		A	297.8	120
			106 (MISS FAST		A	17.7	10
			127 (ADJUST)		A	17.3	5
			190 (CRACKED)		A	67.4	9
			246 (FAULT MAINT)		A	3.6	2
			540 (PUNCTURED)		A	1.0	1
			730 (LOOSE)		A	15.0	1
			750 (MISSING)		A	3.0	2
			780 (BENT)		A	27.0	6
			846 (DELAM)		A	50.6	4
			865 (PROT. COAT)		A	14.0	1
			947 (TORN)		A	26.6	4
					B	559.7	174
		L ADJUST	020 (WORN)		A	2.0	1
			092 (MISMATCH)		A	.2	1
			105 (DAM FAST)		A	.7	2
			106 (MISS FAST)		A	.6	1
			730 (LOOSE)		A	.5	1
					B	4.0	6
		P (REMOVED)	800 (NO DEFECT)		A	45.5	5
			947 (TORN)		A	5.2	1
					B	50.7	6
		Q (INSTL)	190 (CRACKED)		A	3.2	1
			799 (NO DEFECT)		A	19.0	2

Table XIX (Con't.)

66-1 VARIABLE INQUIRY REPORT

SUFFIX	WUC	ACTION	HOW MAL		MANHOURS	UNITS
FV F-111A	11BAB SKIN	Q INSTL	800			
			800 (NO DEFECT)	A	45.0	3
				B	67.2	6
		R REM/REPL	150 (CHATTER)	A	.5	1
			947 (TORN)	A	33.0	2
				B	33.5	3
		S REM/REINSTL	800 (NO DEFECT)	A	25.6	6
				B	25.6	6
		X T-1-S	799 (NO DEFECT)	A	.3	1
				B	.3	1
		Z CORROSION	170 (CORRODED)	A	15.0	2
				B	15.0	2
		9	190 (CRACKED)	A	.3	1
				B	.3	1
				C	821.5	210
	11BAL COVERS	F	020 WORN	A	2.0	1
			105 DAM FAST	A	2.3	1
			127 ADJUST	A	5.8	5
			190 CRACKED	A	129.1	12
			660 STRIPPED	A	3.5	3
			730 LOOSE	A	6.0	1
			799 NO DEFECT	A	3.0	1
			947 TORN	A	55.2	2
				B	206.9	26
		G	020 WORN	A	42.8	7
			070 BROKEN	A	10.6	5
			103	A	1.1	1
			105 DAM FAST	A	915.0	448
			106 MISS FAST	A	68.3	33
			108 BROKEN	A	3.0	1
			117 DETERIOR	A	16.0	4

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Table XIX (Con't.)

66-1 VARIABLE INQUIRY REPORT

SUFFIX	WUC	ACTION	HOW MAL		MAN-HOURS	UNITS
FV F-111A	118AL CovERS	R	106 106	A	50.0	5
			190	A	19.0	4
			800	A	17.9	4
			846	A	.8	1
			947	A	.6	1
				B	138.1	23
		S	800	A	171.7	64
				B	171.7	64
		T	799	A	3.6	3
				B	3.6	3
		U	799	A	3.5	4
				B	3.5	4
		X	799	A	25.3	10
				B	25.3	10
				C	2,261.9	931
	11800 WING GENERAL	G R/R MINOR PARTS	020 (WORN)	A	.3	1
			105 (DAM FAST)	A	7.5	9
			127 (ADJUST)	A	1.0	1
			246 (FAULT MAINT)	A	2.7	7
			805	A	3.0	1
			947 (TORN)	A	2.0	1
				B	16.5	20
		L ADJUST	020 (WORN)	A	1.5	5
			105 (DAM FAST)	A	1.0	1
				B	2.5	6
		Q INSTL	799 (NO DEFECT)	A	15.0	2
				B	15.0	2
		R REM/REPL	105 (DAM FAST)	A	1.0	1
				B	1.0	1
		X T-1-S	799 (NO DEFECT)	A	21.4	6
				B	21.4	6

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Table XIX (Con't.)

66-1 VARIABLE INQJIRY REPORT

SUFFIX	WUC	ACTION	HOW MAL		MANHOURS	UNITS
FV F-111A	11800 WING GENERAL	Y, TSHOOT	190 CRACKED	A	12.0	1
			381 LEAK	A	12.0	1
				B	24.0	2
				C	80.4	37
				D	3,847.5	1,309

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Table XIX (Cont.)

55-1 VARIABLE INQUIRY REPORT

SUFFIX	WUC	ACTION	HOW MAL		MANHOURS	UNITS
F5 (F-111D)	11BAB (SKIN)	G (REPAIR)	105 (DAM FAST)	A	4.0	1
			865 PROT COAT	A	2.0	1
				B	6.0	2
				C	6.0	2
	11BAL COVERS	G	105	A	5.6	3
			106	A	.3	8
				B	5.9	11
		P	800	A	1.0	2
				B	1.0	2
		Q	800	A	1.0	2
				B	1.0	2
				C	7.9	15
				D	13.9	17

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Table XIX (Con't.)

66-1 VARIABLE INQUIRY REPORT

SUFFIX	WUC	ACTION	HOW MAL		MANHOURS	UNITS
F6 F-111E	11BA4 (FRAME)	F (REPAIR)	020 (WORN)	A	8.0	1
				B	8.0	1
		G (R/R MINOR PARTS)	020 (WORN)	A	4.0	2
			070 (BROKEN)	A	2.0	1
			105 (DAM FAST)	A	63.1	9
			190 (CRACKED)	A	1.3	1
				B	70.4	13
		L (ADJUST)	105 (DAM FAST)	A	.5	1
				B	.5	1
		R (REM/REPL)	070 (BROKEN)	A	126.0	2
				B	126.0	2
		V (CLEAN)	230 (CONTAM)	A	.4	2
				B	.4	2
				C	205.3	19
	11BA3 (SKIN)	F (REPAIR)	105 (DAM FAST)	A	9.0	2
				B	9.0	2
		G (R/R MINOR PARTS)	020 (WORN)	A	.5	1
			070 (BROKEN)	A	5.6	1
			105 (DAM FAST)	A	45.3	17
			106 (MISS FAST)	A	16.6	6
			117 (DETERIOR)	A	48.0	1
			190 (CRACKED)	A	6.0	2
			780 (BENT)	A	19.0	2
			846 (DELAM)	A	25.4	2
				B	166.4	32
		X T-1-S	799 (NO DEFECT)	A	2.6	2
				B	2.6	2
				C	178.0	36
	~~11BAL Covers~~	~~F~~	~~105~~	~~A~~	~~9.5~~	~~2~~
			~~190~~	~~A~~	~~17.6~~	
			~~947~~	~~A~~	~~25.4~~	~~2~~
				~~B~~	~~52.5~~	~~4~~

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Table XIX (Con't.)

66-1 VARIABLE INQJIRY REPORT

SUFFIX	WUC	ACTION	HOW MAL		MANHOURS	UNITS
F6	11BAL	G	020			
F-111C	COVERS			A	5.0	1
			105			
				A	195.3	59
			106			
				A	23.0	10
			108			
				A	.8	1
			190			
				A	29.6	3
			865			
				A	1.0	1
			947			
				A	6.0	1
				B	260.7	86
		L	105			
				A	.5	2
			127			
				A	3.0	1
				B	3.5	3
		P	800			
				A	22.4	15
				B	22.4	15
		Q	800			
				A	15.9	6
				B	15.9	6
		S	800			
				A	76.5	18
				B	76.5	18
		T	799			
				A	1.5	2
				B	1.5	2
		J	799			
				A	1.9	2
				B	1.9	2
		V	230			
				A	.2	1
				B	.2	1
		X	170			
				A	2.0	1
				B	2.0	1
				C	437.1	138
	11B0J (WING GENERAL)	G (F/R MINOR PARTS)	105 (DAM FAST)			
				A	1.0	1
			127 (ADJUST)			
				A	.5	1
			947			
				A	5.0	1
				B	6.5	3
		X	799			
		F1-5		A	49.5	26
				B	49.5	26
				C	56.0	29
				D	876.4	222

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Table XIX (Con't.)

65-1 VARIABLE INQUIRY REPORT

SUFFIX	WUC	ACTION	HOW MAL			MAN-HOURS	UNITS
F7	~~11BAL~~	A	780	12W7925127			
F-111F	~~COVERS~~				A	6.0	1
					B	6.0	1
			105		A	68.2	14
			106		A	31.6	7
			135		A	1.0	1
			190		A	16.6	1
					B	119.4	23
		P	660		A	.8	1
			800		A	6.0	1
					B	6.8	2
		Q	800		A	1.4	1
					B	1.4	1
		R	070		A	2.0	1
			105		A	1.3	1
					B	3.3	2
					C	136.9	29
	11BOO	X	690		A	4.0	1
	WG GENERAL	TEST-INSP-SERV	VIBRATION		B	4.0	1
					C	4.0	1
					D	140.9	30

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Table XIX (Con't.)

66-1 VARIABLE INQUIRY REPORT

JUNE - NOV 1972

SUFFIX D	WUC C	ACTION B	HOW MAL A			MANHOURS	UNITS
FV (F-111A)	11BAA FRAME	F (REPAIR)	190 (CRACK)	12B5848/1	A	3.0	1
					B	3.0	1
		G (R/R MINOR PARTS)	105 (FASTENER)		A	4.0	4
			106 (FASTENER)		A	19.0	5
			135 (BINDING)		A	4.0	2
			190 (CRACKED)		A	28.0	4
			425 (NICKED)		A	5.2	3
			780 (BENT)		A	2.0	1
					B	62.2	19
					C	65.2	20
	11BAB (SKIN)	F (REPAIR)	106 (FASTENER)		A	3.0	1
					B	3.0	1
		G (R/R MINOR PARTS)	105 (FASTENER)		A	13.2	7
			117 (DETERIORATED)		A	6.0	1
			190 (CRACKED)		A	3.0	1
			846 (DELAMINATED)		A	61.0	5
					B	83.2	14
		L (ADJUST)	105 (FASTENER)		A	3.5	3
					B	3.5	3
		P (REMOVED)	800 NO DEFECT		A	2.6	2
					B	2.6	2
		R (REMOVE/REPLACE)	947 (TORN)		A	.5	1
					B	.5	1
		S (REMOVE/REINSTL)	800 (NO DEFECT)		A	1.6	1
					B	1.6	1
		X (TEST-INSPECT)	799 (NO DEFECT)		A	.3	1
					B	.3	1
					C	94.7	23
	~~11BAL~~	~~F~~	~~105~~		~~A~~	~~2.0~~	~~1~~
			~~127~~		~~A~~	~~4.0~~	~~1~~
			~~190~~		~~A~~	~~28.9~~	~~5~~
					~~B~~	~~35.5~~	~~7~~

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Table XIX (Con't.)

66-1 VARIABLE INQUIRY REPORT

SUFFIX	WUC	ACTION	HOW MAL		MANHOURS	UNITS
F5 F-111D	11BAA FRAME	G R/R MINOR PARTS	190 CRACKED	A	4.0	1
				B	4.0	1
				C	4.0	1
	11BAB SKIN	G R/R MINOR PARTS	105 FASTENER	A	1.5	1
			106 (FASTENER)	A	11.0	2
			117 (DETERIORATED)	A	6.0	3
			540 (PUNCTURED)	A	5.0	1
			780 (BENT)	A	2.0	1
			865 (PROT COATING)	A	4.0	2
				B	29.5	10
				C	29.5	10
	11BAL ACCESS COVER	G R/R MINOR PARTS	105 FAST	A	37.5	19
			106 FAST	A	7.0	3
			108 WIRE OR KEY	A	.3	1
			190 CRACKED	A	8.0	1
			750 MISSING	A	.3	2
			780 BENT	A	1.6	1
			865 PROT COATING	A	4.0	1
				B	54.3	28
		L ADJUST	127 OUT ADJUST	A	1.0	1
				B	1.0	1
		R REM/REPL	105 FAST	A	1.0	1
				B	1.0	1
		S REM/REINSTL	800 NO DEFECT	A	4.5	2
				B	4.5	2
				C	60.8	32
	11BAF LDG EDGE	G R/R MINOR PARTS	105 FAST	A	23.5	8
			106 FAST	A	2.0	1
			108 WIRE OR KEY	A	.5	1
			190 CRACKED	A	7.0	
			947	A	.6	1
				B	33.5	11

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Table XIX (Con't.)

66-1 VARIABLE INQUIRY REPORT

SUFFIX	WUC	ACTION	HOW MAL		MANHOURS	UNITS
F5 F[illegible]	11BAP LDG EDGES	S REIN/ADJUST	800 NO DEFECT	A	2.[illegible]	1
				B	2.[illegible]	1
				C	35.[illegible]	12
	11B00 WING	G (R/R MINOR PART)	106 FAST	A	2.0	1
				B	2.0	1
				C	2.0	1
				D	131.9	56

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Table XIX (Con't)

66-1 VARIABLE INQUIRY REPORT

SUFFIX	WUC	ACTION	HOW MAL		MANHOURS	UNITS
F6 F-111E	118AB SKIN	G R/REMOVE PART	190 CRACKED	A	3.0	1
				B	3.0	1
				C	3.0	1
	118AL ACCESS COVERS	G/ R/MINOR PARTS	105 FRST	A	8.8	[illegible]
			10_ FAST	A	1.6	1
			127 OUT ADJST	A	2.0	1
			750 MISSING	A	.3	3
			846 DELAM	A	2.0	1
				B	14.7	13
				C	14.7	13
				D	17.7	14

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Table XIX (Con't.)

66-1 VARIABLE INQUIRY REPORT

SUFFIX	WUC	ACTION	HOW MAL		MANHOURS	UNITS
FT	11BAA	G	105			
FT	11BAA	G	105 FASTENER	A	4.8	1
F-111F	FRAME	R/R MINOR PARTS	106 FASTENER	A	8.0	1
			117 DETERIORATED	A	3.0	7
			190 CRACKED	A	2.0	1
			780 BENT	A	4.0	1
			947 TORN	A	9.0	2
				B	30.8	13
				C	30.8	13
	11BAB SKIN	G R/R MINOR PARTS	108 WIRE/KEY	A	3.0	2
				B	3.0	2
		R REM/REPL	108 WIRE/KEY	A	4.0	1
				B	4.0	1
				C	7.0	3
	11BAL Access Covers	G R/R Minor Parts	020 Worn	A	2.0	1
			105 Fast	A	53.0	16
			106 Fast	A	19.4	6
			730 Loose	A	3.5	1
			947 Torn	A	2.0	1
				B	80.0	25
		P Removed	105 Fast	A	.5	1
				B	.5	1
		Q Instl	799 No Defect	A	2.0	1
				B	2.0	1
		R Rem/Repl	108 Wire/Key	A	27.5	2
				B	27.5	2
		S Rem/Reinst	800 No Defect	A	21.0	4
				B	21.0	4
				C	131.2	33
	11BAP Lug Edges	G R/R Minor Parts	105 Fast	A	9.9	[illegible]
			106 Fast	A	1.8	[illegible]
			108 Wire/Key	A	.3	1

D470753

Table XIX (Con't)

66-1 VARIABLE INQUIRY REPORT

SUFFIX	WUC	ACTION	HOW MAL		MANHOURS	UNITS
F7 F-111F	118AP LOG EDGES	G R/R MINOR PARTS	127 OUT ADJUST	A	2.6	1
				B	14.6	5
				C	14.6	5
	11800 WING	G R/R MINOR PARTS	105 FAST	A	11.0	1
			780 BENT	A	3.0	1
				B	14.0	2
		H EQUIP CHKO	799 NO DEFECT	A	.4	1
				B	.4	1
		Y TROUBSHOOT	799 NO DEFECT	A	6.0	1
				B	6.0	1
				C	20.4	4
				D	204.0	58

Table XX
WORK UNIT CODE

WORK UNIT CODE	
11000	*AIRFRAME
	CENTER SECTION DOORS (CONT)
11AED	DOOR, GROUND COOLING AND SERVICE AIR CONNECTION (2202)
11AEE	DOOR, ENGINE START (3321)
11AEF	DOOR, HYDRAULIC DISCONNECT (3334)
11AEH	HINGE, DOOR HYD DISCONNECT
11AE9	NOC
	AFT SECTION
11AFA	FRAME
11AFB	BULKHEAD
11AFC	SKIN
11AFD	FAIRING
11AFE	LINERS, INSULATING
11AFF	SHIELDS, HEAT
11AFG	SUPPORT ASSY, SHOCK STRUT MLG
11AFH	DORSAL ASSY, AFT
11AFJ	CONE ASSY, SPEED BUMP
11AFK	SUPPORT, ENGINE MOUNT
11AFL	SUPPORT, MLG ACTUATING CYL
11AFM	SUPPORT ASSY
11AFN	TUBE ASSY, TORQUE, ARRESTOR HOOK
11AFP	CENTERBODY FRAME
11AFQ	FAIRING, TAIL BUMPER
11AFT	PIN, SHEAR, UPPER REMOVABLE FRAME
11AF9	NOC
	AFT SECTION COVERS
11AGA	COVER, FUSELAGE UPPER AFT (4411) (4412)
11AGB	COVER, FUSELAGE UPPER AFT (4425) (4426)
11AGC	COVER, FUSELAGE UPPER AFT (4437) (4438)
11AGD	COVER, STABILIZER ACTUATOR (4445) (4446)
11AGE	COVER, FUSELAGE AFT ACCESS (4447) (4448)
11AGF	COVER, FWD ENGINE ACCESS (4121) (4221)
11AGH	COVERS, MISC
11AGJ	COVER CENTERBODY, UPPER (4460)
11AGK	COVER THROUGH DECK, UPPER (4451) (4452)
11AGL	COVER, STAB SHOULDER (4107) (4207)
11AGM	COVER LOWER STAB SHOULDER (4109) (4209) (4111) (4211)
11AGN	COVER LOWER FUSELAGE (4320)

WORK UNIT CODE	
11000	*AIRFRAME
	AFT SECTION COVERS (CONT)
11AGP	COVER CENTERBODY, BOTTOM (4340)
11AGQ	COVER CHAFF DISPENSER (4113) (4213)
11AGR	COVER FWD CHAFF DISPENSER (4112) (4212)
11AG9	NOC
	AFT SECTION DOORS
11AHA	DOOR, FWD ENGINE (4101) (4201)
11AHB	STRAKE, FWD ENGINE DOOR
11AHC	DOOR, MID ENGINE UPPER (4104) (4204)
11AHD	STRAKE, MID ENGINE DOOR
11AHE	DOOR, AFT ENGINE (4105) (4205)
11AHF	STRAKE, AFT ENGINE DOOR
11AHG	DOOR, LOWER MID ENGINE ACCESS (4311) (4312)
11AHH	LATCH, SIDE, FWD ENGINE DOOR
11AHN	LATCH, MID ENGINE DOOR
11AHP	DEVICE, HOLD-OPEN, FWD ENGINE DOOR
11AHQ	FAIRING, HINGED, SAFETY LATCH
11AHR	DOOR ACCESS, PNEUMATIC (4345)
11AHS	DOOR, STAB SHOULDER, HINGED (4106) (4206)
11AHU	LATCH, SAFETY, FWD ENGINE DOOR
11AHV	CAPSULE, SAFETY LATCH, FWD ENGINE DOOR
11AH9	NOC
11AJ0	VERTICAL STABILIZER
11AJA	FRAME
11AJC	FAIRING
11AJD	LEADING EDGE
11AJE	TIP
11AJF	COVERS, ACCESS
11AJG	DOOR, ACCESS
11AJ9	NOC
11B00	WINGS
	GENERAL
11BAA	FRAME
11BAC	LEADING EDGE, FIXED STUB
11BAD	LEADING EDGE, SECTION 1
11BAE	LEADING EDGE, SECTION 2
11BAF	LEADING EDGE, SECTION 3
11BAG	LEADING EDGE, SECTION 4
11BAH	TIP, WING
11BAK	TRAILING EDGE
11BAL	COVERS, ACCESS
11BAB	SKIN

WORK UNIT CODE	
11000	*AIRFRAME
	GENERAL (CONT)
11BAM	PIN ASSY, WING PIVOT
11BAN	BEARING ASSY, WING PIVOT
11BAP	LEADING EDGES, WING AND PYLON ACCESS
11BA9	NOC
	PYLON PIVOT CONTROL MECHANISM
11BBA	TUBE ASSY, TIE ROD
11BBB	LEVER ASSY, PIVOT PYLON
11BBC	HOUSING, ROTATING PIVOT PYLON
11BBD	BEARING, PIVOT PYLON LOWER
11BBE	BEARING, PIVOT PYLON UPPER
11BBF	SEAL ASSY, FUEL DRAG LINK
11BBG	TUBE ASSY, DRAG LINK
11BB9	NOC

Table XXI
MALFUNCTION CODE

HOW MALFUNCTIONED CODES NUMERIC LISTING

Code	How Malfunctioned
001	Gassy
003	Open Filament or Tube Circuit
004	Low GM or Emission
007	Arcing, Arced
008	Noisy
009	Microphonic
010	Poor or Incorrect Focus
020	Worn, Chafed, or Frayed
025	Capacitance Incorrect
028	Conductance Incorrect
029	Current Incorrect
033	Destroyed or Removed From Service as a Result of Testing
037	Fluctuates, Unstable or Erratic
051	Fails to Tune or Drifts
064	Incorrect Modulation
065	High Voltage Standing Wave Ratio
069	Flame Out
070	Broken
080	Burned Out or Defective Lamp, Meter, or Indicating Device
086	Improper Handling
088	Incorrect Gain
092	Mismatched - Wheel Halves, Electronic Parts, etc
094	No Gain or Emission
103	Attack Display Malfunction
105	Loose or Damaged Bolts, Nuts, Screws, Rivets, Fasteners, Clamps, or Other Common Hardware
106	Missing Bolts, Nuts, Screws, Rivets, Fasteners, Clamps, or Other Common Hardware
108	Broken, Faulty, or Missing Safety Wire or Key
111	Burst or Ruptured
116	Cut
117	Deteriorated
127	Adjustment or Alignment Improper
130	Change of Value
135	Binding, Stuck, or Jammed
142	Engine Removed - Excessive Maintenance
150	Chattering
158	Launch Damage
160	Contacts/Connection Defective
167	Torque Incorrect
169	Incorrect Voltage
170	Corroded -- Mild to Moderate
177	Fuel Flow Incorrect
190	Cracked
204	Accidental, explosion of, or damage from on board munitions
230	Dirty, Contaminated, or Saturated by Foreign Material

HOW MALFUNCTIONED CODES (Cont)

Code	How Malfunctioned
242	Failed to Operate or Function - Specific Reason Unknown
246	Improper or Faulty Maintenance
253	Misfires
255	No Output/Incorrect Output
277	Fuel Nozzle Coking
279	Spray Pattern Defective
290	Fails Diagnostic/Automatic Test
300	Grounded Electrically
301	Foreign Object Damage
303	Bird Strike Damage
314	Slow Acceleration
315	RPM Fluctuation or Incorrect
317	Hot Start
330	Excessive Hum
334	Temperature Incorrect
350	Insulation Breakdown
372	Metal on Magnetic Plug
374	Internal Failure
380	Compressor Damaged - Reason Unknown
381	Leaking - Internal or External
382	Liquid Lock
383	Lock on Malfunction
386	Maintenance Action Due to Lost in Flight Occurrence
398	Oil Consumption Excessive
410	Lack of, or Improper Lubrication
424	External Power Source
425	Nicked
447	Wrong Logic
450	Open
454	Open Trace -- Multilayer Printed Circuit Board (Depot use only)
457	Oscillating
458	Out of Balance
464	Overspeed
469	Bushing Worn or Damaged
472	Fuse Blown or Defective Circuit Protector
475	Engine Failed to Start
481	Keyway or Spline Damaged or Worn
486	Turbine Damaged - Reason Unknown
503	Sudden Stop
513	Compressor Stall
518	Improper Routing
520	Pitted
525	Pressure Incorrect
537	Low Power or Thrust
540	Punctured
553	Does Not Meet Specification, Drawing, or Other Conformance Requirements (Must be used with "When Discovered" Code Y)
561	Unable to Adjust to Limits

HOW MALFUNCTIONED CODES (Cont)

Code	How Malfunctioned
567	Resistance Incorrect
583	Scope Presentation Incorrect or Faulty
585	Sheared
599	Travel or Extension Incorrect
602	Failed or Damaged Due to Malfunction of Associated Equipment or Item
605	Crazed
606	Counter Run Off-Position Indicator
607	No-Go Indication - Specific Reason Unknown
615	Shorted
617	Sulfidation
619	Shimmy Excessive
622	Wet/Condensation
623	After-Burner Blowout
624	After-Burner No-Light
625	Gating Incorrect
626	Inductance Incorrect
627	Attenuation Incorrect
629	Relay Malfunction - Replacement Not Required
632	Expended (Thermal Battery, Fire Extinguisher, etc)
635	Sensitivity Incorrect
636	Time Delay Incorrect
637	Triggering Incorrect
649	Sweep Malfunction
651	Air in System
652	Automatic Align Time Excessive
653	Ground Speed Error Excessive
654	Terminal Error - CEP Excessive
655	Terminal Error - Range Excessive
656	Terminal Error - Azimuth Excessive
657	Distance Measurement Error-Navigation Equipment
658	Bearing Destination (Station) Error
660	Stripped
664	Tension Incorrect
667	Corroded -- Severe
668	Pump Isolation - Possible Damaged Due to Lack of Fluid
669	Potting Material Melting (Reversion Process)
690	Vibration Excessive
692	Video Faulty
693	Audio Faulty
694	Audio and Video Faulty
695	Sync Absent or Incorrect
697	Faulty Tape - Program or Checkout
698	Faulty Card - Program or Checkout
709	Administrative Condemnation
710	Bearing Failure or Faulty
711	Improper Blanking
719	Broken or Frayed Bonding or Ground Wire
730	Loose
731	Battle Damage
748	Frequency Erratic or Incorrect
750	Missing

Table XXI (Cont.)
MALFUNCTION CODE

HOW MALFUNCTIONED CODES (Cont)	
780	Bent, Buckled, Collapsed, Dented, Distorted, or Twisted
782	Tire Tread Area Defective - Use Cut, Delaminated, Punctured, Worn, etc, if applicable
783	Tire Sidewall Damaged or Defective
784	Tire Bead Area Damaged or Defective
785	Tire Inside Surface Damaged or Defective
793	No Defect - TCTO Kit Received by Base Supply or Parts Are Available in Supply
797	No Defect - Technical Order Previously Complied With
798	No Defect - Technical Order Not Applicable - Equipment to be Replaced, Modified, or Not Installed
799	No Defect
800	No Defect - Component Removed and/or Reinstalled to Facilitate Other Maintenance
801	No Defect - Technical Order Compliance
802	No Defect - Partial Technical Order Compliance
803	No Defect - Removed for Time Change
804	No Defect - Removed for scheduled Maintenance or Modification
805	No Defect - Not Otherwise Coded-Electrical, Hydraulic, Air, Fuel, Oxygen, Oil, etc., Lines, Mechanical Linkage and Cables. Servicing a Component Removed/Disconnected and/or Installed/Connected to Facilitate Other Maintenance
812	No Defect - Indicated Defect Caused by Associated Equipment Malfunction
816	Impedance Incorrect
824	Gyro Processes
838	B Plus Incorrect
846	Delaminated
865	Protective Coating Sealant Missing or Defective
877	Transportation Damage
878	Weather Damage
884	Lead Broken
900	Burned or Overheated
901	Intermittent
910	Chipped
911	Engine TCTO Correction Code (Reference T.O. 00-20-4)

HOW MALFUNCTIONED CODES (Cont)	
916	Impending or Incipient Failure Indicated by Spectrometric Oil Analysis
917	Impending Failure or Latent Defect Indicated by Non-Destructive Inspection (Do not use if other codes apply)
931	Accidental or Inadvertent Operation, Release, or Activation (Use Code 386 if item was lost in flight)
932	Does Not Engage, Lock, or Unlock Correctly
935	Scored or Scratched
937	Overheated Cathode Stem
938	Power Output Dip
939	Unable to Load Program
941	Non-Programmed Halt
942	Illegal Operation or Address
943	Data Error
944	Parity Error
946	Incorrect or No Print Out
947	Torn
948	No Defect-Operator Error
949	Computer Memory Error/Defect
955	Data Link High Error Rate
956	Abnormal Function of Computer Mechanical Equipment
957	No Display
958	Incorrect Display
959	Fails to Transfer to Redundant Equipment
961	High Anode Current
962	Low Power - Electronic
963	Broken Filament/Cathode Terminal
964	Poor Spectrum
966	RF Wind Suck-in, Broken or Cracked
968	Dioding
969	Cannot Resonate Input Cavity
971	Cracked Cathode Bushing
972	Damaged Input Probe
973	Damaged Output Probe
974	Does Not Track Tuning Curve
975	Filament to Cathode Short
981	Frequency Instability
982	Frozen Tuning Mechanism
983	Grid to Cathode Short
984	Grid to Plate Short
985	High Body Current/Beam Interception
986	High Modulator Inverse
987	Input Pulse Distortion
988	Loss of Vacuum
990	No Focus Current
991	Out of Band Frequency
992	Output Pulse Distortion
993	RF Drive Improper

HOW MALFUNCTIONED CODES (Cont)	
994	RF Feed-Thru Attenuated/Distorted
995	RF Feed-Thru Completely Interrupted
996	RF Terminal Overheated
997	RF Window Burned

Table XXII
REPAIR ACTION CODE

ACTION TAKEN CODES

A BENCH CHECKED AND REPAIRED. This code will be entered when bench check and repair of any one item is accomplished at the same time. (Also see Code F.)

B BENCH CHECKED - SERVICEABLE (No Repair Required). This code will be entered when the item is bench checked and no repair was required.

C BENCH CHECKED - REPAIR DEFERRED. This code will be entered when bench check is accomplished and repair action is deferred. (See Code F.)

D BENCH CHECKED-TRANSFERRED TO ANOTHER BASE OR UNIT. Item is bench checked at a forward operating base, dispersed operating base or enroute base and is found unserviceable and transferred to a main operating base or home base for repair. This code will not be used for items returned to a depot for overhaul. This code will also be used when PME or other equipment is sent to another base or unit for bench check, calibration, or repair and is to be returned; and for items forwarded to contractors on base level contracts.

1. BENCH CHECKED - NRTS (Not Reparable This Station) - REPAIR NOT AUTHORIZED. This code will be entered when the shop is not authorized to accomplish the repair. This code shall only be used when the repair required to return an item to serviceable status is specifically prohibited by current technical directives. This code shall not be used due to lack of authority for equipment, tools, facilities, skills, parts, or technical data.

BENCH CHECKED - NRTS - LACK OF EQUIPMENT, TOOLS, OR FACILITIES. This code will be entered when the repair

ACTION TAKEN CODES (Cont)

is authorized but cannot be accomplished due to lack of equipment, tools, or facilities. This code shall be used without regard as to whether the equipment, tools, or facilities are authorized or unauthorized.

3. BENCH CHECKED - NRTS - LACK OF TECHNICAL SKILLS. This code will be entered when repair cannot be accomplished due to lack of technically qualified people.

4. BENCH CHECKED - NRTS - LACK OF PARTS. This code will be entered when parts are not available to accomplish repair.

5. BENCH CHECKED - NRTS - SHOP BACKLOG. This code will be entered when repair cannot be accomplished due to excessive shop backlog.

6. BENCH CHECKED - NRTS - LACK OF TECHNICAL DATA. This code will be entered when repair cannot be accomplished due to lack of maintenance manuals, drawings, etc, which describe detailed repair procedures and requirements.

7. BENCH CHECKED - NRTS - EXCESS TO BASE REQUIREMENTS. This code will be entered when repair will not be scheduled for shop repair due to item being excess to base requirements.

8. BENCH CHECKED - Returned to Depot-Return to Depots by direction of System Manager (SM) or Item Manager (IM). Use only when items that are authorized for base level repair are directed to be returned to depot facilities by specific written or verbal communication from the IM or SM: or when items are to be returned to depot facilities for modification in accordance with a Time Compliance Technical Order (TCTO), or as UR exhibits.

ACTION TAKEN CODES (Cont)

9. BENCH CHECKED - CONDEMNED. This code will be entered when the item cannot be repaired and is to be processed for condemnation, reclamation, or salvage. This code will also be used when a "condemned" condition is discovered during field maintenance disassembly or repair.

E INITIAL INSTALLATION. This code will be used for installation actions that are not related to a previous removal action such as installation of additional equipment or installation of an item to remedy a ship-short condition. This code will be used only for equipment managed under the Advance Configuration Management System. Reference T.O.'s 00-20-2-3, 00-20-2-5, and 00-20-2-7. Must be used with How Mal Code 799.

F REPAIR. This code will not be used to code "on-equipment" work if another code will apply. When it is used in a shop environment, this code will denote repair as a separate unit of work after a bench check. Shop repair includes the total repair man-hours and includes cleaning, disassembly, inspection, adjustment, reassembly, and lubrication of minor components incident to the repair when these services are performed by the same work center. For precision measurement equipment this code will be used only when calibration of the repaired item is required. (See code G).

G REPAIR AND/OR REPLACEMENT OF MINOR PARTS, HARDWARE, AND SOFTGOODS (Seals, Gaskets, Electrical Connectors, Fittings, Tubing, Hose, Wiring, Fasteners, Vibration Isolators, Brackets, etc). Work unit codes do not cover most nonreparable items. Therefore, when items such as those identified above are repaired or replaced, this action taken code will be used. When this action taken code is used, the work unit code will identify the assembly being serviced or most directly related to parts being repaired or replaced. For example, if an electrical connector was repaired and was attached

Table XXII (Cont.)
REPAIR ACTION CODE

ACTION TAKEN CODES (Cont)

to a radio transmitter, the work unit code for the transmitter would be used with this action taken code. For precision measurement equipment, this code will be used for repairs that do not require calibration of the repaired item. (See code F.)

H EQUIPMENT CHECKED - NO REPAIR REQUIRED (For "On-Equipment" Work Only). This code will be used for all discrepancies which are checked and found to require no further maintenance action. This code will be used only if it is definitely determined that a reported deficiency does not exist or cannot be duplicated. Must be used with How Mal Code 799, 812, or 948.

J CALIBRATED - NO ADJUSTMENT REQUIRED. Use this code when an item is calibrated and found serviceable without need for adjustment or is found to be in tolerance but is adjusted merely to peak or maximize the reading. If the item requires adjustment to actually meet calibration standards or to bring in tolerance, use Code K.

K CALIBRATED - ADJUSTMENT REQUIRED. Use this code when an item must be adjusted to bring it in tolerance or meet calibration standards. If the item was repaired or needs repair in addition to calibration and adjustment, use Code F.

L ADJUST. Includes tighten, adjust, bleed, balance, rig, fit, or actuating reset button or switch. Enter this code whenever a particular discrepancy is cleared by adjusting, etc, the item. If the identified component also requires replacement bits and pieces as well as adjustment (new points, condenser, tubes, etc), enter the appropriate repair code instead of L.

M DISASSEMBLE. This code will be entered for disassembly action when the complete maintenance job is broken into parts and reported as such. Do not use for On Equipment Work.

ACTION TAKEN CODES (Cont)

N ASSEMBLE. This code will be entered for assembly action when the complete maintenance job is broken into parts and reported as such. Do not use for On Equipment work.

P REMOVED. This code will be entered when an item is removed and only the removal is to be accounted for. In this instance, delayed or additional actions will be accounted for separately. (Also see codes Q, R, S, T, and U.) Do not use for Off Equipment work.

Q INSTALLED. This code will be entered when an item is installed and only the installation action is to be accounted for. (Also see codes E, P, R, S, T, and U.) Do not use for Off Equipment work.

R REMOVE AND REPLACE. This code will be entered when an item is removed and another like item is installed. (Also see codes T and U.) Do not use for Off Equipment work.

S REMOVE AND REINSTALL. This code will be entered when an item is removed, for any reason, and the same item reinstalled. (Also see codes T and U.) Do not use for Off Equipment work. Must be used with How Mal Code 800, 804, or 805.

T REMOVED FOR CANNIBALIZATION. This code will be entered when a component is cannibalized. The work unit code will identify the component being cannibalized. Do not use this code for Off Equipment work. Must be used with How Mal Code 799.

U REPLACED AFTER CANNIBALIZATION. This code will be entered when a component is replaced after cannibalization. Do not use this code for Off Equipment work. Must be used with How Mal Code 799.

V CLEAN. This code will be entered when cleaning is accomplished to correct a discrepancy and/or when cleaning is not accounted for as part of a repair action, such

ACTION TAKEN CODES (Cont)

as code F. Includes washing, acid bath, buffing, sand blasting, degreasing, decontamination, etc. Cleaning and washing of complete items, such as ground equipment, vehicles, missiles, or airplanes,should be recorded by utilizing support general codes.

X TEST-INSPECT-SERVICE. This code will be entered when an item is tested or inspected or serviced (other than bench check) and no repair is required. This code does not include servicing or inspection chargeable to support general work unit codes.

Y TROUBLESHOOT. Enter this code when time expended in locating a discrepancy is great enough to warrant separating the troubleshoot time from the repair time. Use of this code necessitates completion of two separate line entries or two separate forms: one for the troubleshoot phase and one for the repair phase. When recording the troubleshoot time separate from the repair time, the total time taken to isolate the primary cause of the discrepancy should be recorded, utilizing the work unit code of the defective subsystem or system. Do not use for Off Equipment work.

Z CORROSION REPAIR. Includes cleaning, treating, priming, and painting of corroded items. This code should always be used when actually treating corroded items, either on equipment or in the shop. The work unit code should identify the item that is corroded. Use support general code for painting or corrosion preventive treatment prior to an item becoming corroded.

Because of the basic need to perform in-service inspection on the F-111, the necessary data to discuss items (a) thru (f) of paragraph 3.1.1.1.8.1 of FZP-1402 Addendum 1. is not available. For a limited discussion of items (a) thru (f), F-111 Production experience is used. Since in-service repair of F-111 wing structure is a depot level task, the following discussion is considered to be typical to a depot maintenance approach:

(a) Frequency and type of Structural Inspections:

See Table XVIII

(b) Accessibility Factors: The Wing Box structure has limited accessibility for inspection without removal of SPAR access covers and/or the top wing skin. The latter approach gives access into the inner structure of the wing box where investigation or repair of a problem is necessary. Removal of the top wing skin is a major task.

(c) NDI techniques used: The basic inspection of the wing box is limited to visual. X-ray of sub-structure parts has been accomplished on completed wings in order to locate manufacturing induced cracks. This task was highly specialized and is not considered adequate for general inspection. X-ray inspection is more applicable with the top wing skin removed.

(d) Demonstrated and assumed damage size detection capability: Generally damage size detectable is that related to a warped skin surface or loose or missing fastener since the inspection is visual. Fuel leaks may further enhance the capability to visually find flaws.

(e) Typical morphology (shape or structure) and location of damage found in the various types of inspection: Figure 80 shows the typical type of sub-structure damage located within an assembled wing structure using X-ray. The cracks so located were production induced cracks for which the general location in the wing was known.

(f) Identification as to detection of single and multiple cracks: Limited experience with the F-111 wing in this area other than as discussed in (e).

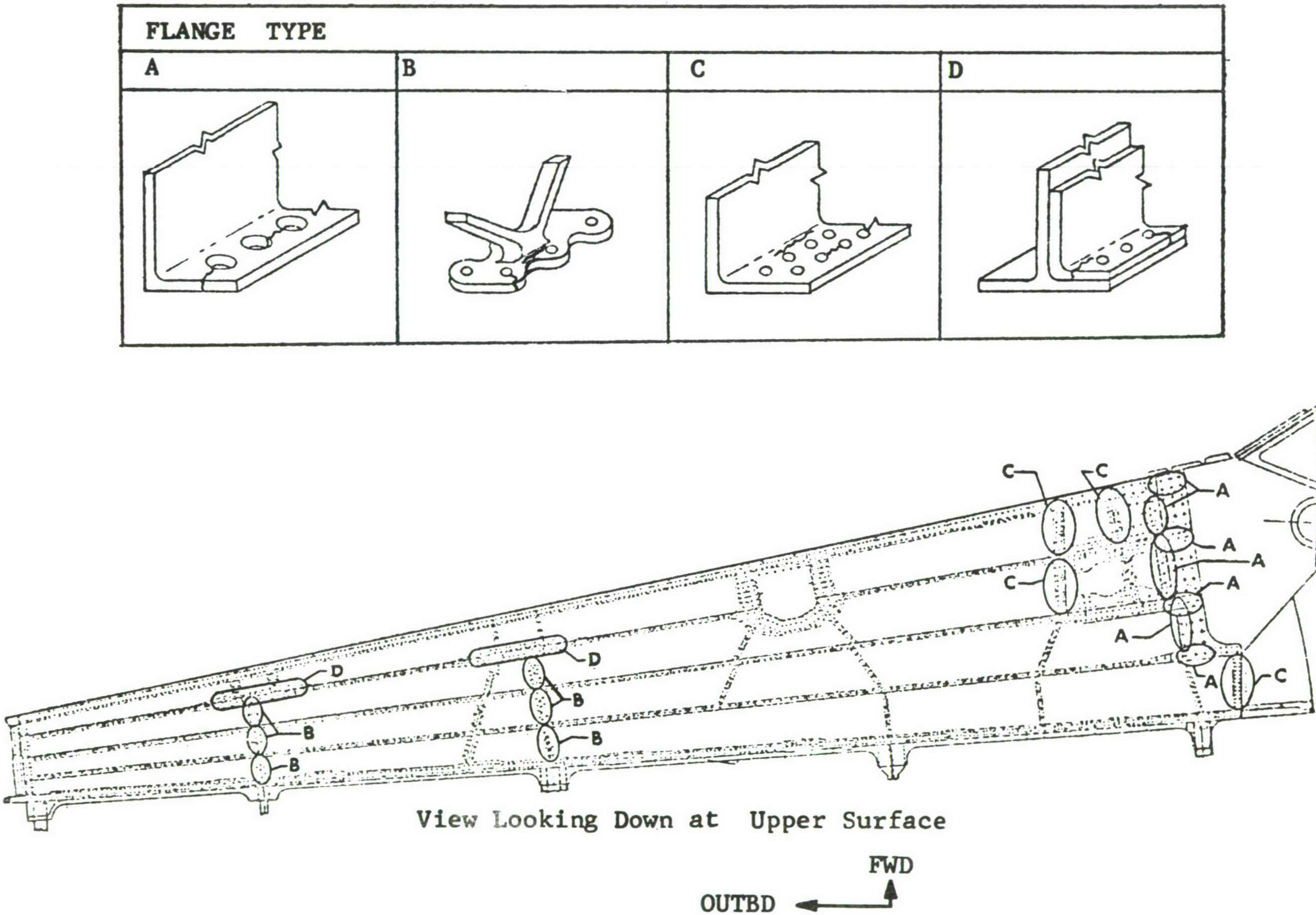


Figure 80 Typical Type of Cracks in Wing Substructure

IX.3.9.1.1 Baseline Fatigue Test Program Inspections. Two full scale wing components, designated as A-4, have been fatigue tested in the F-111 fatigue test program; one right hand wing and one left hand wing. Each test article experienced catastrophic failure outside the defined baseline, which is, the wing box only, outboard of the wing-to-wing pivot fitting splice.

A review of the inspection activity performed during these tests indicates that no "in-service" problems were discovered in the wing box itself. The greatest degree of inspection was directed at the D6ac steel wing pivot fitting and not the wing box. Because of this, no attempt will be made in this study to detail the inspection activity for each wing tested. However, to typically illustrate test inspections, a description of scheduled inspections performed on the right hand wing will be given followed by a brief summary of special inspections performed.

IX.3.9.1.2 Inspections - A-4 Right Hand Wing Component. Three levels of scheduled inspections (complete, major, minor) were performed on the test specimen. Complete inspections were performed at the ends of Blocks 10, 20, and 30. Major inspections were performed at the ends of Blocks 3, 7, 13, 17, 23, and 27. Minor inspections were performed at the ends of all other blocks and after Condition T-2 of every block.

A minor inspection consisted of performing a visual inspection of external surfaces of the test specimen with emphasis on a few control points. A major inspection consisted of a minor inspection plus a visual inspection of some internal areas which were accessible through selected doors. All control points except for the lower pivot lug were visually inspected. A complete inspection consisted of a major inspection plus the dye penetrant inspection of selected control points including the lower pivot lug.

After the test had been started, X-ray and Magnetic Rubber Inspection (MRI) requirements were added to the major and complete inspections. X-ray pictures were taken of lower surface fuel flow holes in the pivot fitting and center spar from C.S.S. 140 to 170. The X-ray pictures were then sent to General Dynamics Convair Aerospace Division, Fort Worth operation for evaluation. Except for the major inspection after

Block #3, MRI was performed on the 28 fuel flow holes in the spanwise stiffeners of the pivot fitting lower plate at each major and complete inspection.

During the complete inspection after Block 10, a delta scan ultrasonic inspection was performed on the lower plate of the pivot fitting.

The results of the scheduled inspections which were described above were negative. The results of the special inspections which are described below were negative. Also presented below are:

A. Minor modifications of specimen parts
B. Replacements of specimen parts
C. Actions resulting from discrepancies

IR/AR No.	DESCRIPTION
A130753	The four 10-23 UNF holes in the bottom of pivot pin, 12W415, were enlarged to 1/4-28 UNF. Reference pin rotation mechanism. 9 Dec. 1969.
A210180	Verified that front spar web, 12W477-14, was heat treated to 220-240 KSI. 12 Dec. 1969.
A210170	Upper pivot bushing, 12W426-13, surface scratches were polished. 17 Dec. 1969.
A210195	Reworked accidental damage to inboard auxiliary flap track, 12W8961. 18 Dec. 1969.
A212132	Repaired accidental dent in fixed leading edge 12W6010. 6 Jan. 1970
A212115	Removed surface defect in fuel flow hole #20 of 12W473. 21 Jan. 1970
A212121	Performed delta scan ultrasonic inspection of pivot fitting lower plate, 12W473, 19 Jan. 1970 Replaced lower fiberglass panel, 12W495. 22 Jan. 1970
A212130	Removed surface defect in fuel flow hole #9 of 12W473. 10 Feb. 1970

A212152	Upper pivot bushing, 12W426-13 cracked. Both lower and upper pivot bushings, 12W426-9 and -13 were replaced at the Fort Worth operation. 20 Feb. 1970.
	After Block 27, a 1.0 inch long crack was found in upper pivot bushing, 12W426-13. The test was continued without replacing the bushing. The crack was monitored for the rest of the test, and it did not get larger.
A235422	Pivot fitting lower plate failed in block 30. The test specimen was sent to the Fort Worth operation for failure analysis. 1 May 1970.

IX.3.9.2 Parametric Analysis

The intent of this paragraph was basically to determine the impact of in-service inspections determined in paragraph IX.3.9.1 on the damage tolerance criteria by establishing the design stress for varying inspection intervals and varying initial damage assumptions.

The results of the study reported in paragraph IX.3.9.1 indicate that no in-service inspections are performed on the baseline wing other than visual. However, the data in Figure 81 has been developed to illustrate how accomplishment of the task may be approached. The curves in Figure 81 were obtained by cross-plotting the data in Figure 17 of paragraph IX.3.1 for the part through surface flaw (t = .611) in conjunction with a/0 values based on Q = 2.46 for a semi-circular flaw shape. Additional curves may be developed in a similar manner for other a/Q values and for other flaw types for which data in paragraph IX.3.1 is given.

IX.3.9.3 Variation of Parameters and Multiple Cracking

The range of variations for the parameters used in the parametric analyses of paragraph IX.3.9.2 were to have been determined. However, the in-service inspection survey reported in paragraph IX.3.9.1 shows that only visual inspection is performed on the baseline wing structure thus preventing the development of a parametric analysis. The results of the in-service inspection study also prevented the integration of data from paragraph IX.3.9.1 and IX.3.9.2 in order to evaluate the apsects of multiple cracking.

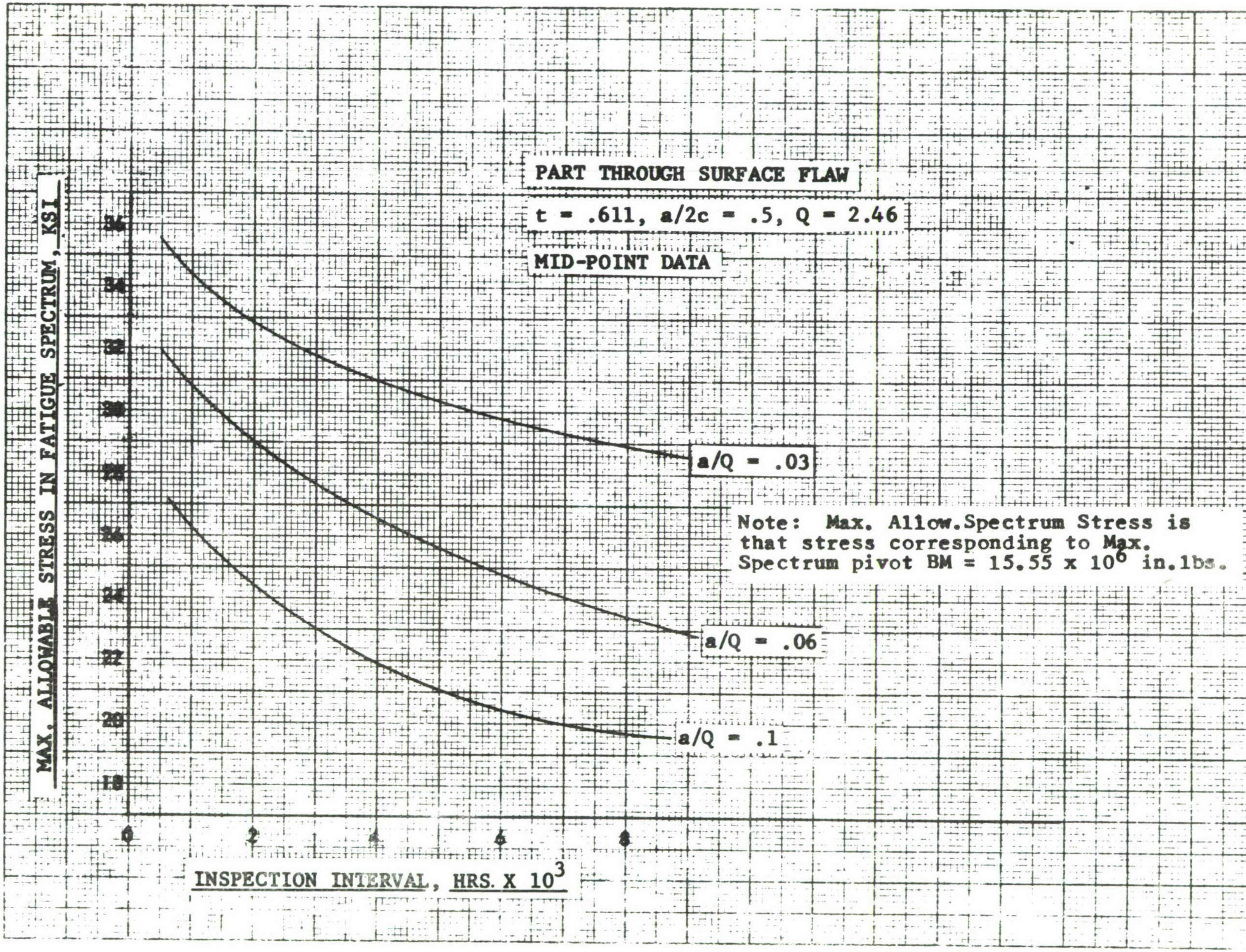


Figure 81 Allowable Stress Level vs. Length of Inspection Interval

IX.3.10 Effects of Varying Residual Strength Load Requirements

The objective of this study is to assess the impact on design allowable stress and life of varying the residual strength load requirement (P_{xx}) from the load that could occur in 100 inspection intervals, to that which could occur in 10 and 1 inspection intervals.

A baseline wing pivot bending moment exceedance distribution was generated for inspection intervals associated with the following degrees of inspectability:

(1) 2.5 hours - representative of a single flight and is associated with the ground evident degree of inspectability.

(2) 25 hours - representative of 10 flights and is associated with the walk around visual degree of inspectability.

(3) 400 hours - representative of 1 calendar year and is associated with the special visual degree of inspectability.

(4) 1000 hours - representative of 1/4 airplane life and is associated with the depot or base level degree of inspectability.

(5) 4000 hours - representative of 1 airplane life and is associated with non inspectability.

The one-time occurrence bending moment level for 100, 10 and 1 times each of the above inspection intervals was established. A complete description of this study is given below.

Crack growth design allowable curves (Max. spectrum stress level versus initial flaw-size) were then developed to typically illustrate the impact resulting from the use of the one-time occurrence loads in determination of design stress. These curves were developed for inspectability classes (2) through (5) described above, and for the part through surface flaw (t = .611") case only.

IX.3.10.1 F-111F Baseline Wing Box Residual Strength Studies

Reference: (a) FZM-12-10783, F-111A/E/D Mission Analysis to Determine Maneuver Load Factor Exceedance Spectra, dated 27 June 1969

(b) FZS-12-168A, F-111A/E/D/F Fatigue Loads Spectra, dated 30 October 1972

(c) MIL-A-8866A, Airplane Strength and Rigidity Reliability Requirements, Repeated Loads, and Fatigue, dated 31 March 1971

(d) MIL-A-8861A, Airplane Strength and Rigidity, Flight Loads, dated 31 March 1971

In support of the ADP wing residual strength sensitivity studies the four enclosed wing pivot net bending moment cumulative frequency distributions were developed. A distribution was generated for each of the following inspection intervals:

(1) 2.5 hours - Ground Evident Inspectability

(2) 25 hours - Walk Around Visual Inspectability

(3) 400 hours - Special Visual Inspectability

(4) 1000 hours - Depot or Base Level Inspectability

(5) 4000 hours - Non-Inspectable

Study results are summarized in Table XXIII and the bending moment exceedance curves are given in Figures 82 through 86 . The most interesting result shown in Table XXIII is that the one time loads derived for a 4000 hour interval were less than those derived for the smaller inspection intervals. This result is directly related to variations in slope of the F-111 load factor exceedance curves in Figures 82 through 86 at the point where these curves are extrapolated down to the one-time level. Extrapolation is accomplished by drawing a straight line through the two highest load levels determined for each inspection interval. The resulting slope for the 4000 hour case is much steeper than that for any other interval as shown in the

Table XXIII

RESIDUAL STRENGTH SUMMARY

INSPECTABILITY	INSPECTION INTERVAL HOURS	ONE TIME LOAD (1) 10^6 IN.LB.
INFLIGHT EVIDENT	1 x 2.5	13.0
	10 x 2.5	15.1
	100 x 2.5	17.3
GROUND EVIDENT	1 x 2.5	13.0
	10 x 2.5	15.1
	100 x 2.5	17.3
WALK-AROUND VISUAL	1 x 25	15.0
	10 x 25	18.3 (2)
	100 x 25	18.3 (2)
SPECIAL VISUAL	1 x 400	15.4
	10 x 400	17.4
	100 x 400	18.3 (2)
DEPOT LEVEL	1 x 1000	15.8
	10 x 1000	17.6
	100 x 1000	18.3 (2)
NON-INSPECTABLE	1 x 4000	15.0
	10 x 4000	15.6
	100 x 4000	16.2

(1) Wing Bending Moment at the Pivot

(2) Max. Moment within V_H and Angle-of-Attack Limits

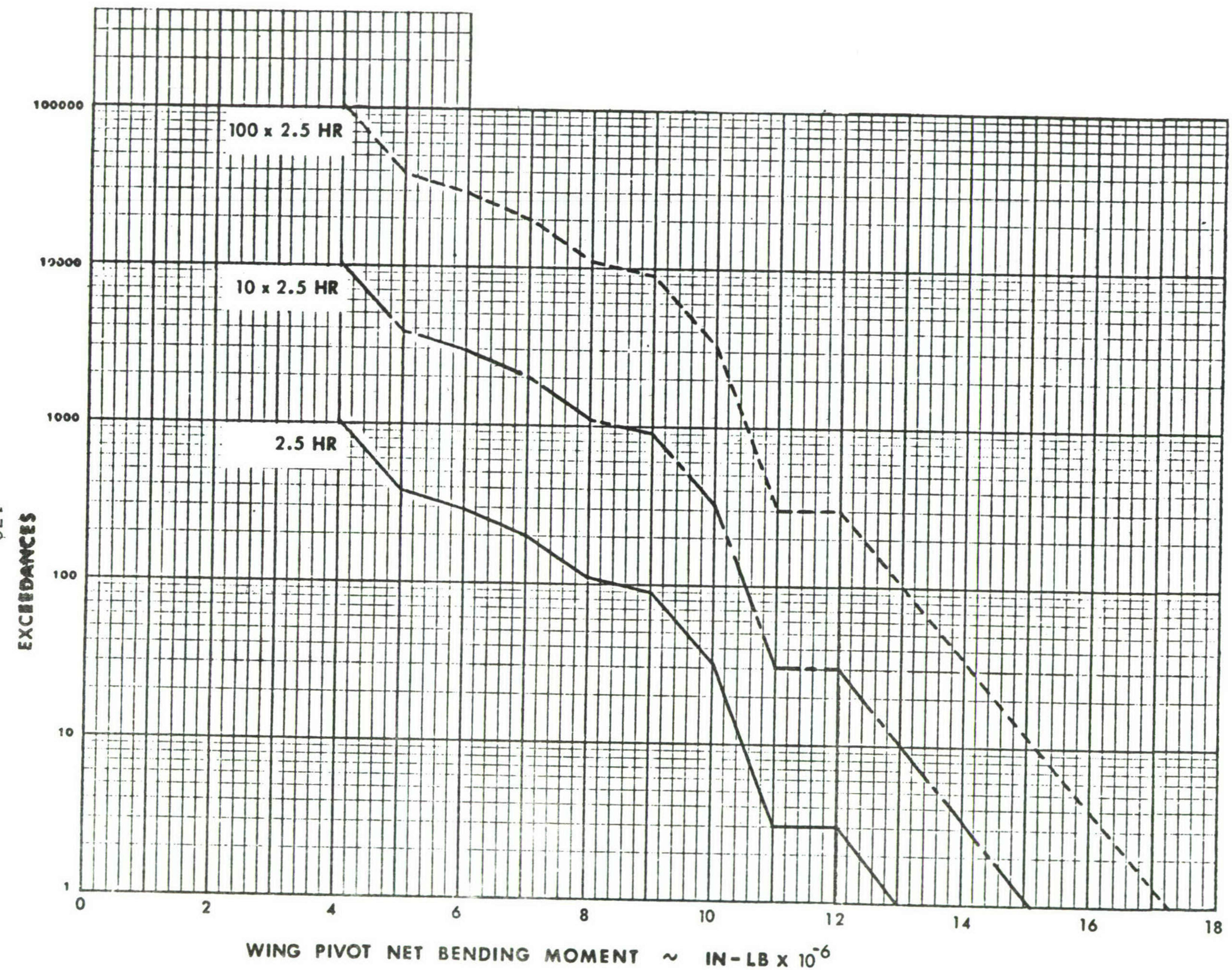


Figure 82 2.5 Hour Inspection Interval

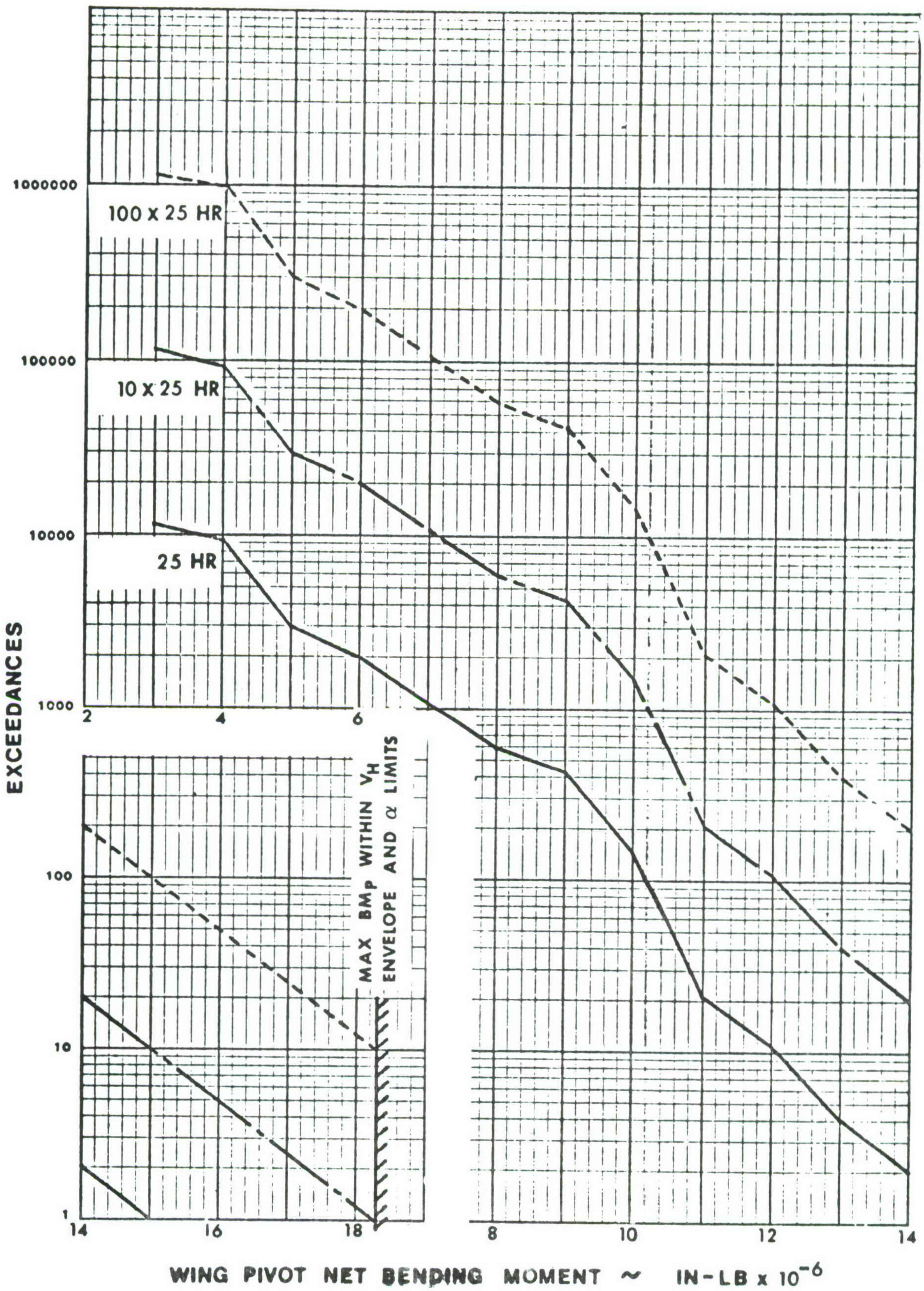


Figure 83 25 Hour Inspection Interval

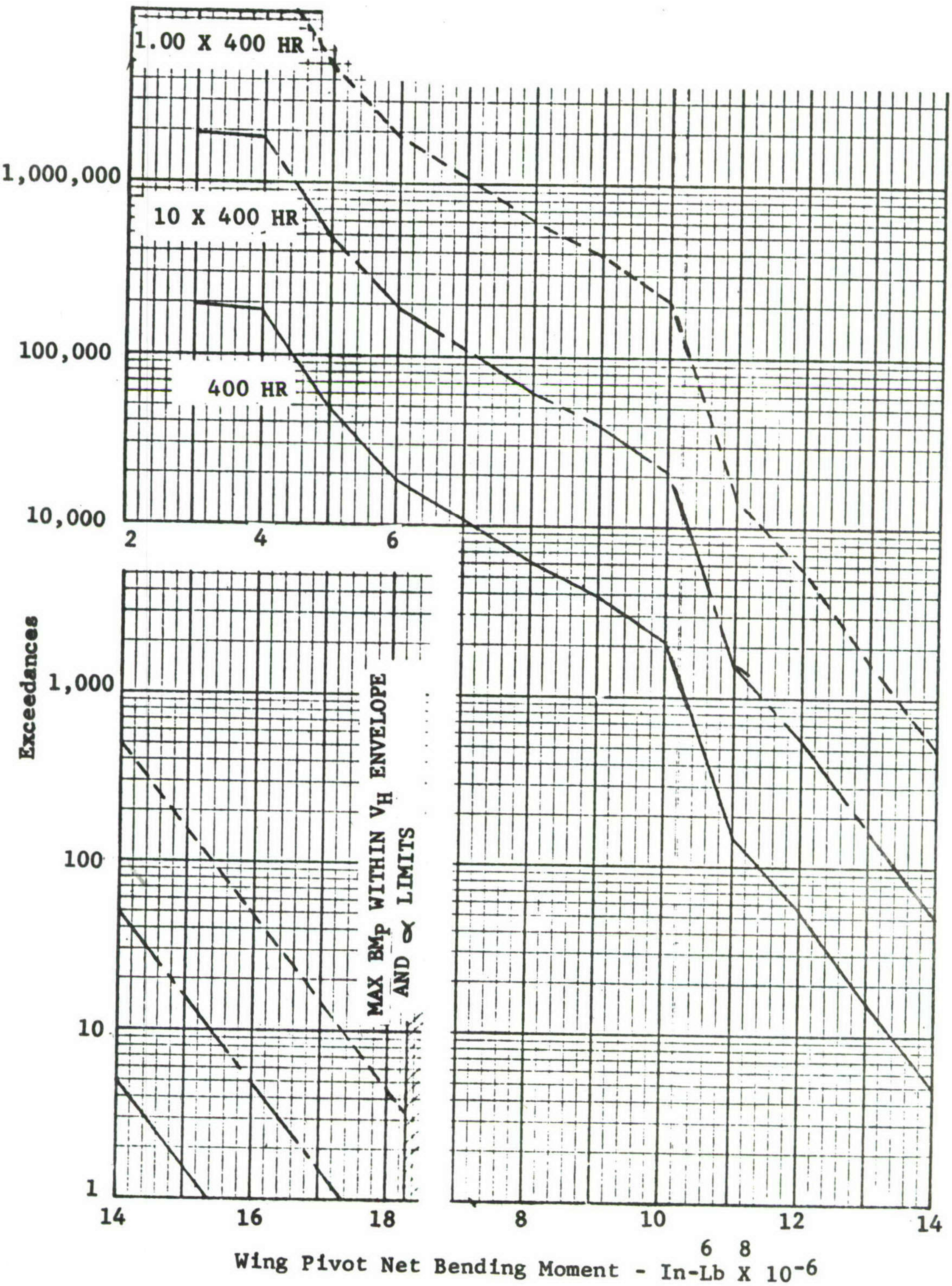


Figure 84 Cumulative Frequency Distributions of Wing Pivot Net Bending Moment for a 400 Hour Inspection Interval

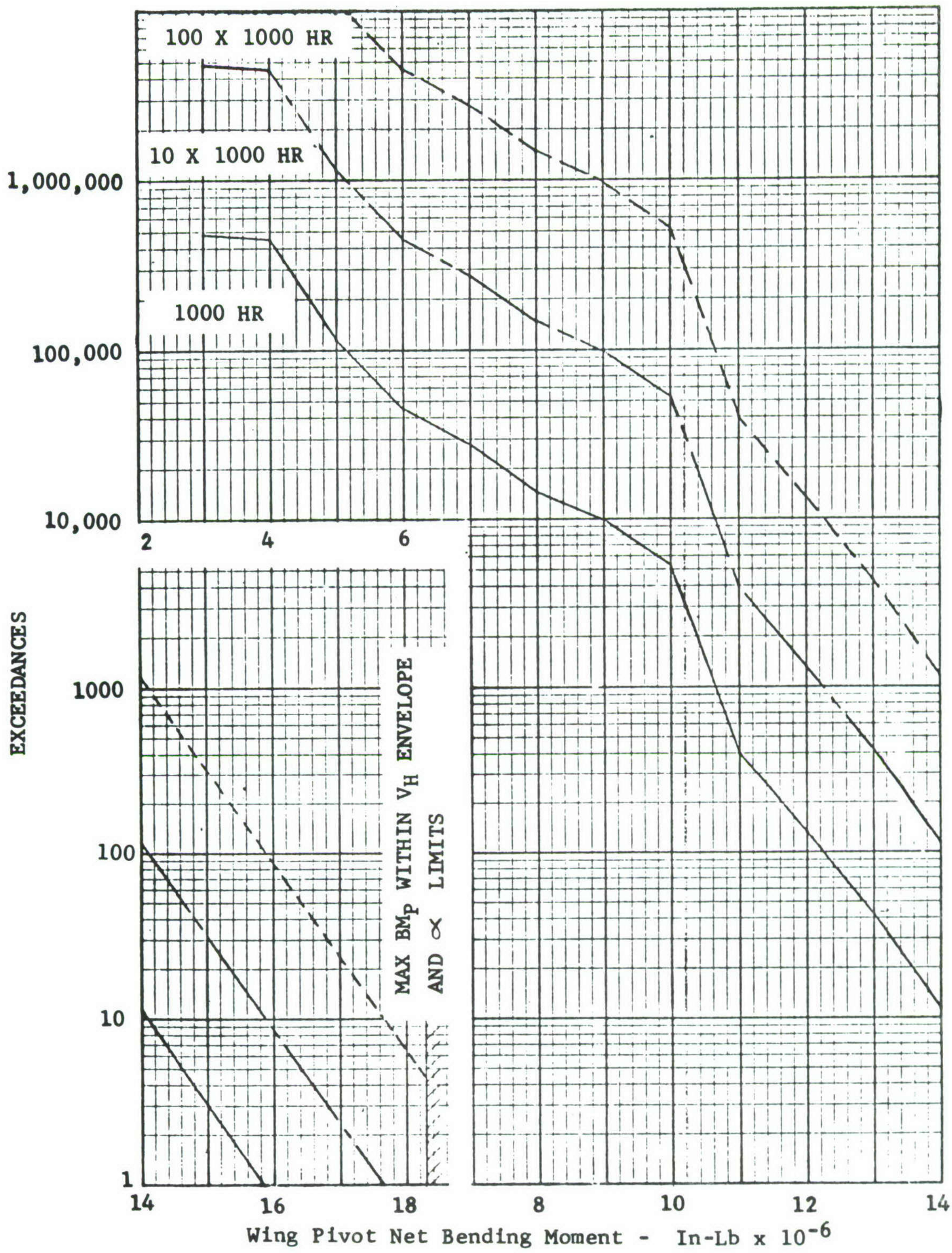


Figure 85 Cumulative Frequency Distributions of Wing Pivot Net Bending Moment for a 1000 Hour Inspection Interval

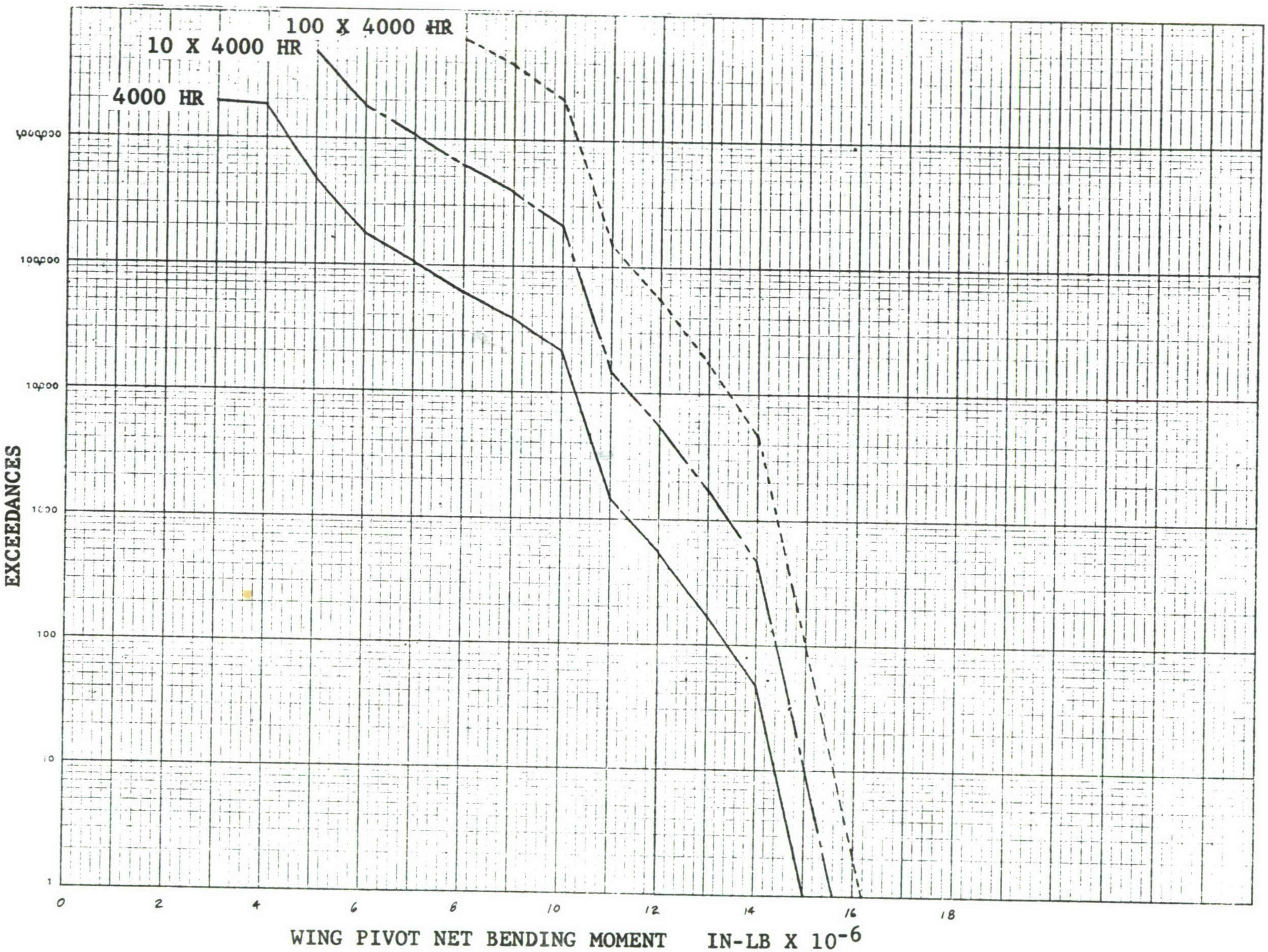
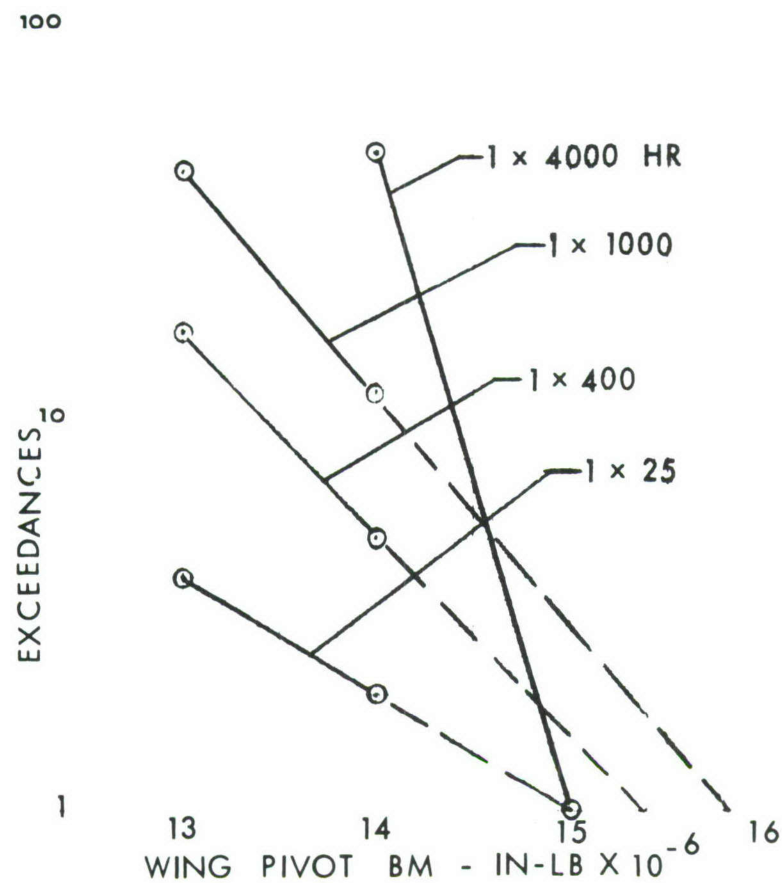


Figure 86 4000 Hr Service Life Cumulative Frequency Distribution Of Wing Pivot Net Bending Moment

sketch below. Development of the exceedance curves are discussed in detail in subsequent paragraphs.



Design allowable curves for a part through surface flaw (t = .611") are presented in Figures 87 through 90. These curves were developed using the one-time occurrence loads in Table XXIII. Table XXIV presents a comparison of design allowable stresses as determined using the curves in Figures 88 through 90 with stresses for the identical flaw case in Figure 17 of paragraph IX.3.1. The curves in Figure 17 were developed using limit load as the residual strength requirement.

The impact of this study on life, for a constant stress level, is illustrated typically by the following example. The current baseline lower skin stress level is 25.2 ksi at the maximum spectrum load. By cross-plotting the data in

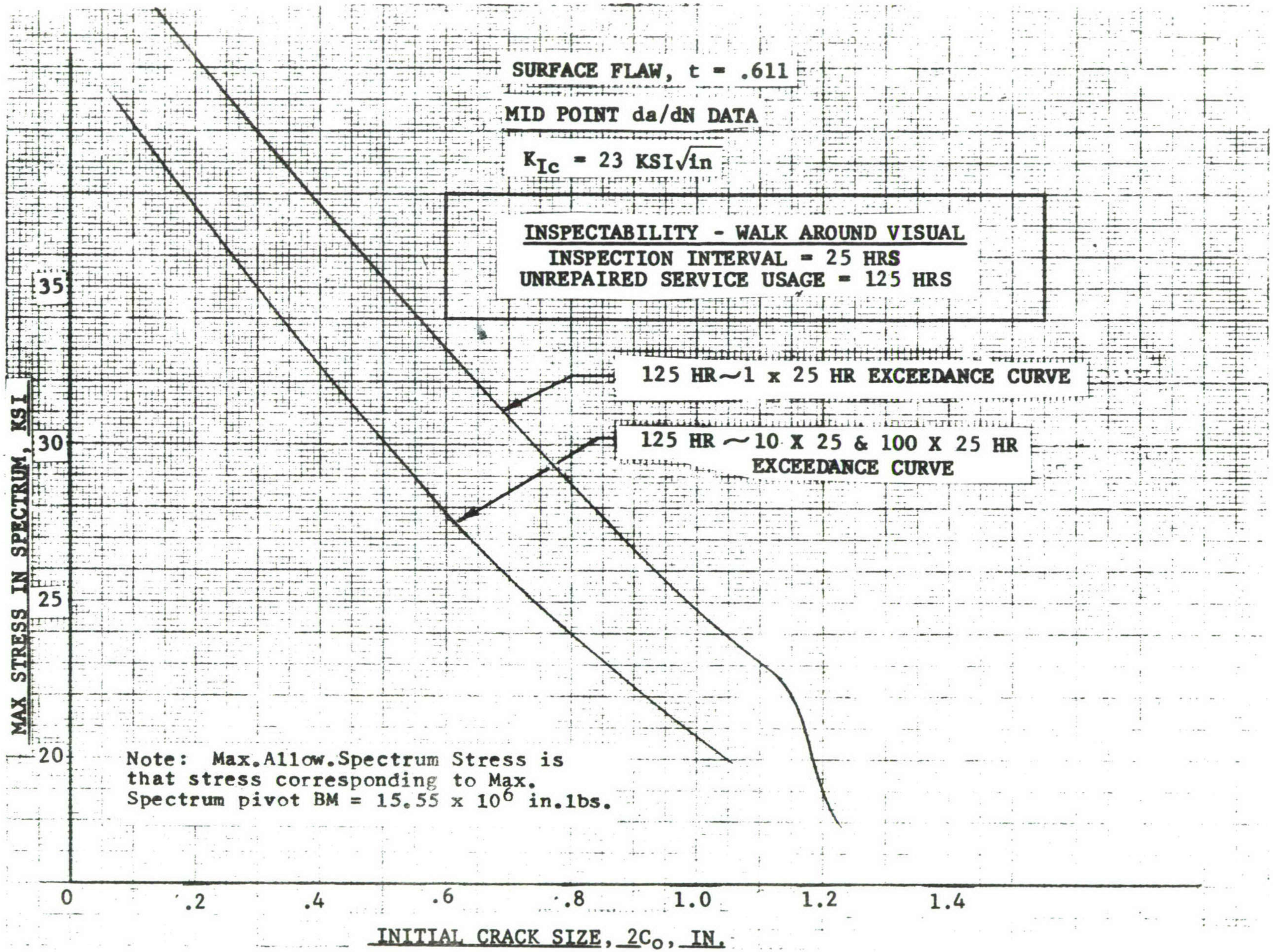


Figure 87 Allowable Curves for 2024-T851 Residual Strength Effects

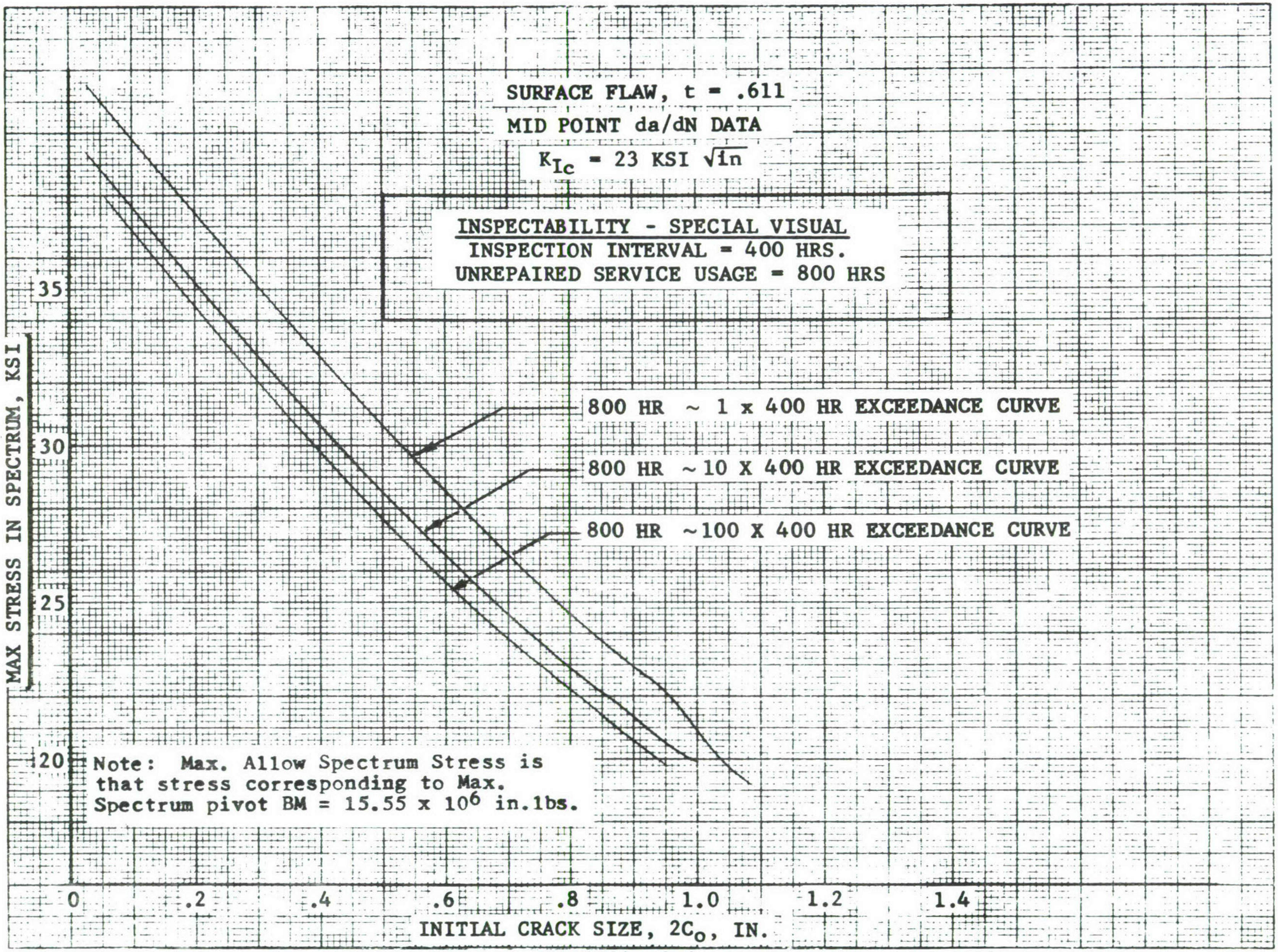


Figure 88 Allowable Curves for 2024-T851 Residual Strength Effects

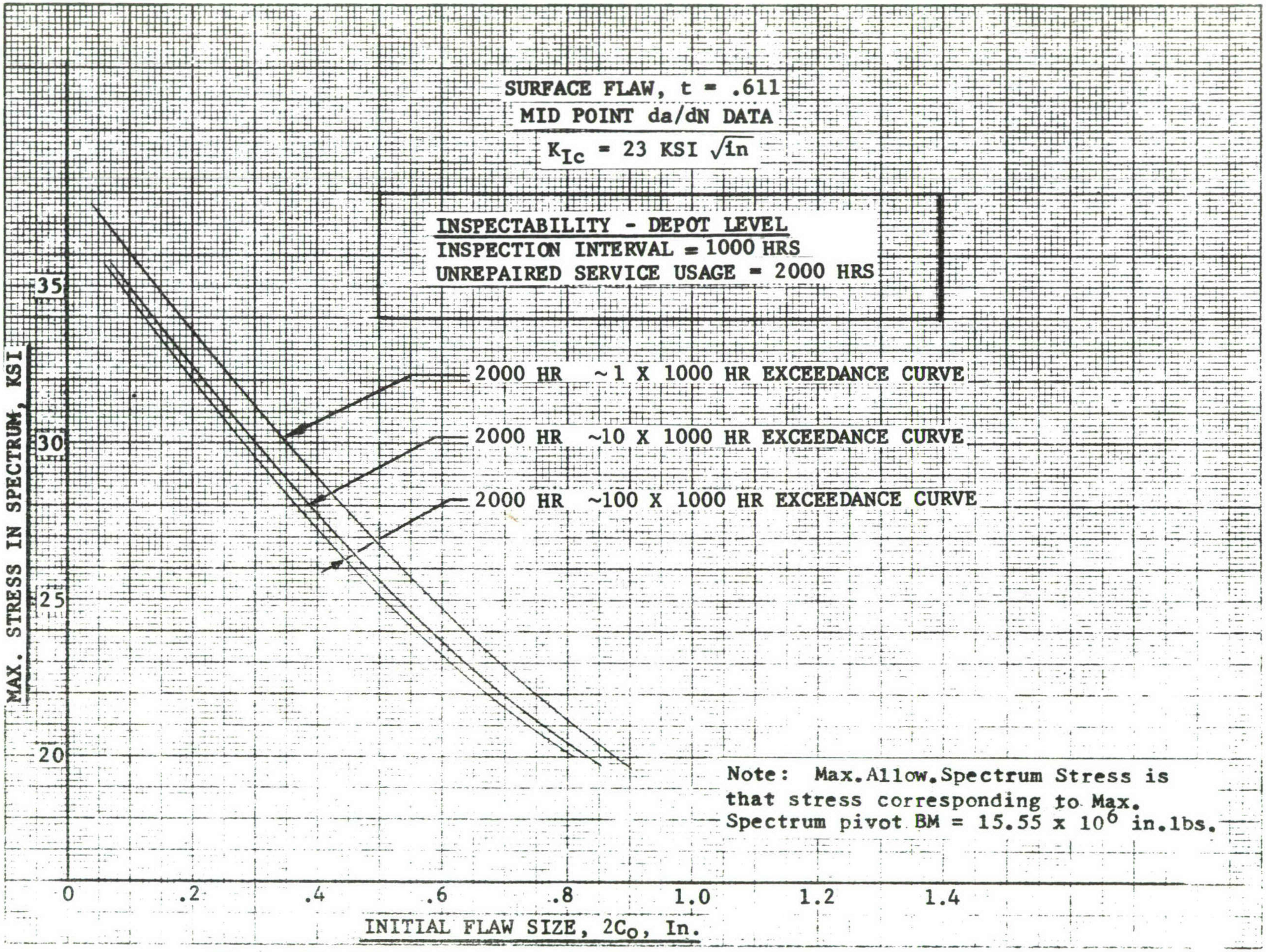


Figure 89 Allowable Curves for 2024-T851 Residual Strength Effects

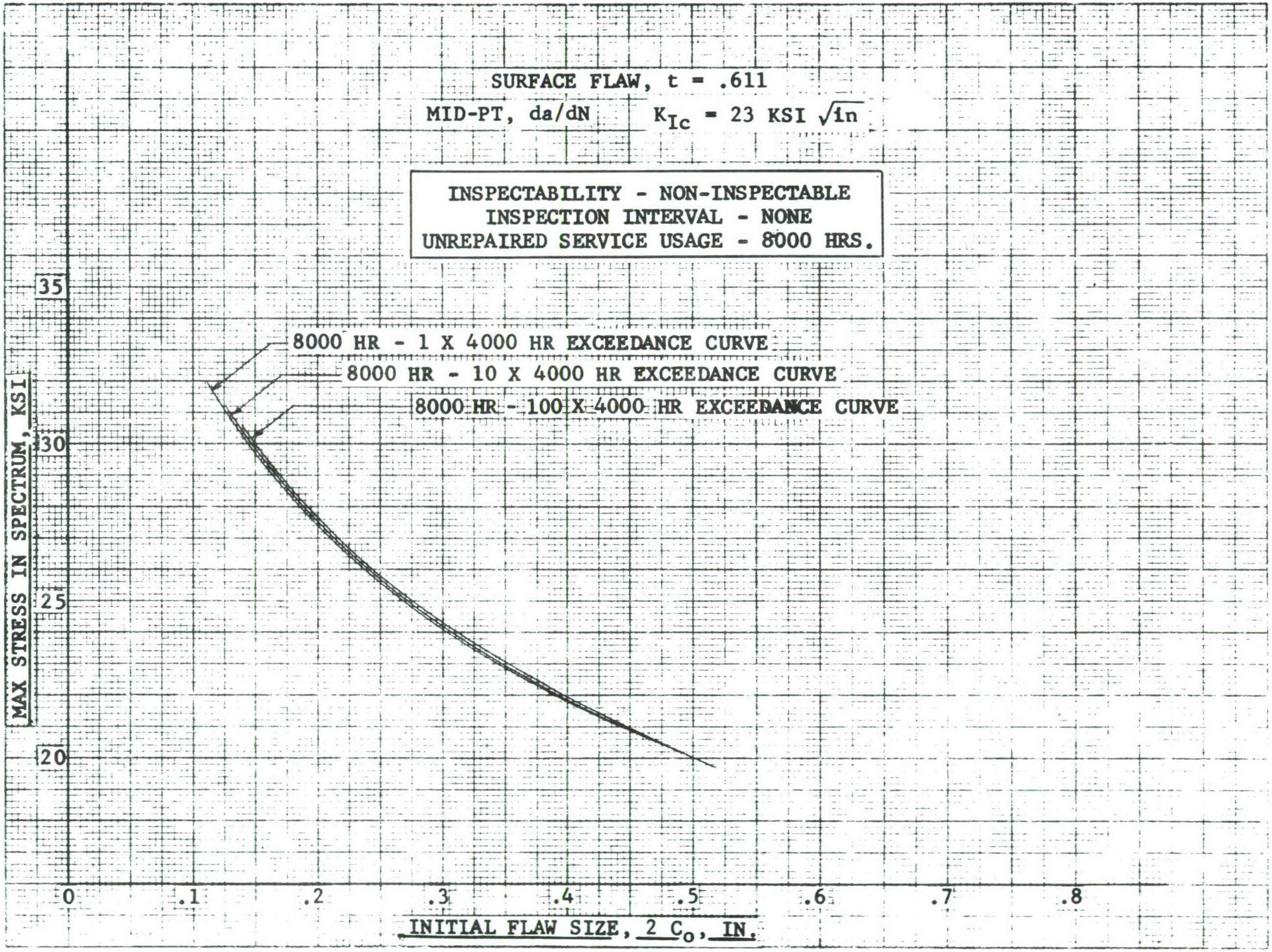


Figure 90 Allowable Curves for 2024-T851 Residual Strength Effects

Table XXIV

IMPACT OF RESIDUAL STRENGTH EFFECT
ON DESIGN ALLOWABLE STRESS LEVEL

2024-T851 Al. F-111 Baseline Severe Usage

Inspectability	Max. Allowable Spectrum Stress Level, ksi			
		Residual Strength Requirement		
	Limit Load	1 x Inspection Interval	10 x Inspection Interval	100 x Inspection Interval
<u>Special Visual</u> o 400-Hr. Inspection Interval o 800-Hr. Unrepaired Usage	26.7	30.9	28.7	27.8
<u>Depot Level</u> o 1000-Hr. Inspection Interval o 2000-Hr. Unrepaired Usage	24.4	26.9	25.8	25.3
<u>Noninspectable</u> o 8000-Hr. Unrepaired Usage	19.7	20.2	20.2	20.2

ANALYSIS CONDITIONS

Surface Flaw--Part Through, $t = 0.611$ in.
$K_{IC} = 23$ ksi$\sqrt{\text{in.}}$, mid-point da/dN, $a/Q = 0.1$
$a/2c = 0.5$, $2c_o = 0.492$ in.

Figures 87 through 90 for this stress, the life versus initial damage curves of Figure 91 were constructed. Entering these curves using the initial flaw size for a/Q = .1 ($2C_o$ = .492"), the life interval for exceedance curve factors of 1, 10, and 100 were determined. The results are 2100 hours (1 factor), 2330 hours (10 factor), and 2730 hours (100 factor). The impact on calculated life using other stress levels or initial flaw sizes may be evaluated in a similar manner.

A detailed discussion of development of the one-time load study is given in the following paragraphs.

The cumulative frequency distributions were based on TAC Phase I and II Training Usage presented in reference (a), wing loads data presented in reference (b), and the normal load factor exceedance data presented in reference (c). Both gust and maneuver spectra were included in the distributions.

The maneuver spectra for each of the inspection intervals were derived from a flight time breakdown to 9 non-TFR conditions, 4 TFR conditions, and 7 high lift conditions.

The gust spectra for each inspection interval classification were derived for the same usage conditions as described for maneuvers; turbulence parameters from reference (d), and F-111 gust response data.

The maneuver spectra for each inspection interval were developed by establishing the normal load factor exceedances that would be indicated by a counting accelerometer for each mission segment (ascent, descent, cruise, loiter, 1000 ft. medium ride TFR, 500 ft. medium ride TFR, 200 ft. hard ride TFR, takeoff and landing). Smaller n_z intervals (1.5 to 2.5 g), not included in previously developed fatigue loads spectra, were added to the inspection interval spectra because of the significant effect these load levels may have on crack growth.

The wing pivot bending moments associated with each usage condition/n_z level in the maneuver loads spectrum were based on the balanced pitch maneuver loads of reference (b), except when α or δ_e limited; in which case unbalanced pitch loads of reference (b) Supplementary Volume 2 were used. For conditions where unbalanced pitch maneuver loads were not available in reference (b), unbalanced pitch loads were derived by multiplying the extrapolated balanced pitch loads from reference (b) by the ratio of available F-111C

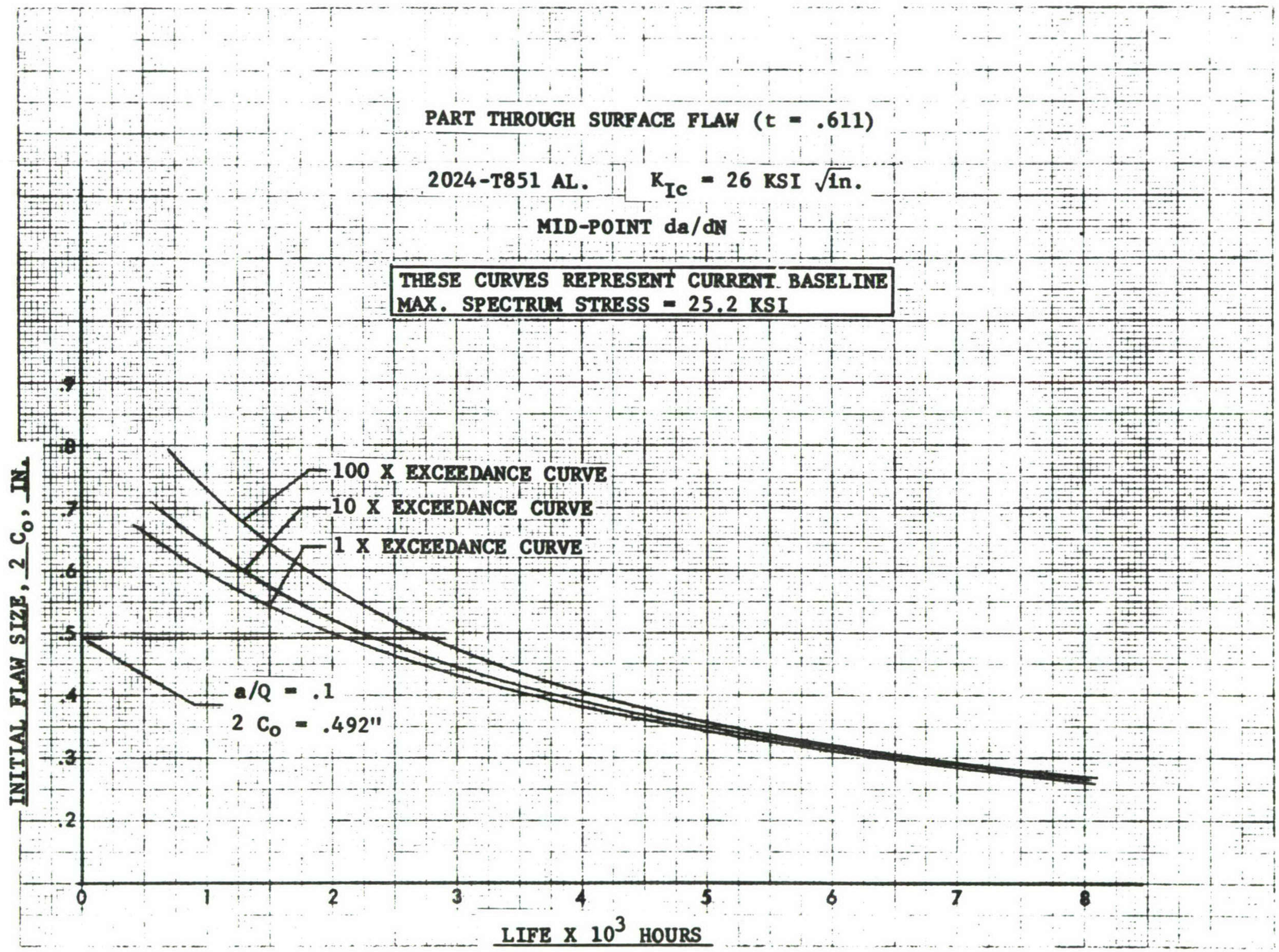


Figure 91 Effect on Life of Constant Allowable Stress

unbalanced pitch to balanced pitch loads at similar usage condition/n_z levels.

2.5 Hour Inspection Interval - The 2.5 hour inspection interval spectrum represents the repeated loads environment that would occur if the WSF(A)-5 Mission Profile of reference (a) were flown one time. This represents the single most severe sortie, in terms of wing pivot bending moment exceedances of the 29 Phase I and II Training Mission Profiles.

The cumulative frequency distribution was the result of the following spectra:

1. Maneuver Loads

   a. Non-TFR Usage
   b. High Lift Usage

2. Gust Loads

   a. Non-TFR Usage
   b. High Lift Usage

The load levels with their associated occurrences were sorted into 1.0×10^6 in-lb increments. The resulting two highest load levels were 13.0×10^6 in-lb and 12.0×10^6 in-lb with cumulative frequencies of 1 and 3, respectively. To obtain the one occurrence levels for 10 and 100 times the 2.5 hour spectrum a straight line extrapolation of the two highest load levels was used.

The maneuver n_z spectra were developed as follows:

(a) Determine the time spent in each mission segment.
(b) Establishing the whole occurrences per mission segment/n_z level from the exceedance data.
(c) Prorating the occurrences to usage conditions for each mission segment/n_z level based on the time spent in each usage condition.
(d) Summing the occurrences over the mission segments.

25 Hour Inspection Interval - The 25 hour inspection interval spectrum is the repeated loads environment that would occur if the following mission profiles were flown one time: WSF(A) - 4 and 5; WSF(P) 2, 3, 6 and 7; and WSF(CR) - 1, 2, 5, and 6. These represent the most severe combination of 10 sorties out of the 29 Phase I and II Training Mission Profiles.

The cumulative frequency distribution was the result of the following spectra:

1. Maneuver Loads

    a. Non-TFR Usage
    b. TFR Usage
    c. High Lift Usage

2. Gust Loads

    a. Non-TFR Usage
    b. TFR Usage
    c. High Lift Usage

The load levels with their associated occurrences were sorted into 1.0×10^6 in-lb increments. The resulting two highest load levels were 14.0×10^6 in-lb and 13.0×10^6 in-lb with cumulative frequencies of 2 and 4, respectively. To obtain the one occurrence load levels for the 25 hour spectrum, the 10 x 25 hour spectrum, and the 100 x 25 hour spectrum a straight line extrapolation of the two highest load levels was used.

The maneuver loads spectrum and the gust spectrum were developed as stated in the previous 2.5 hour spectrum.

400 Hour Inspection Interval - The 400 hour inspection interval spectrum is the repeated loads environment that would occur if the Phase I and II Training Syllabus were repeated approximately 5.5 times. This is representative of 1 year of airplane service.

The cumulative frequency distribution is a result of dividing the typical 4000 hour service life baseline spectrum occurrences by 10. It included the following spectra:

1. Maneuver Loads

    a. Non-TFR Usage
    b. TFR Usage
    c. High Lift Usage

2. Gust Loads

    a. Non-TFR Usage
    b. TFR Usage
    c. High Lift Usage

The 4000 hour baseline spectrum load levels were sorted into 1.0×10^6 in-lb increments and then the associated occurrences were divided by 10. The resulting two highest load levels were 14.0×10^6 in-lb and 13.0×10^6 in-lb with cumulative frequencies of 5 and 17 respectively. To obtain the one occurrence load levels for the 400 hour spectrum, the 10 x 400 hour spectrum, and the 100 x 400 hour spectrum a straight line extrapolation of the two highest load levels was used.

The maneuver n_z spectra were developed as follows:

(a) Determining the time spent in each of the 23 usage blocks per mission segment.

(b) Establishing the fractional occurrences for each usage block per mission segment/n_z level from the exceedance data.

(c) Summing the occurrences over the mission segments.

(d) Prorating the occurrences to maneuver type and rounding off to whole occurrences.

(e) Condensing the 23 usage conditions to 9 and assigning all occurrences to balanced pitch maneuvers.

<u>1000 Hour Inspection Interval</u> - The 1000 hour inspection interval spectrum is the repeated loads environment that would occur if the Phase I and II Training Syllabus were repeated approximately 13.7 times. This is representative of one-fourth a typical airplane service life.

The cumulative frequency distribution is a result of dividing the typical 4000 hour service life baseline spectrum occurrences by 4. It includes the following spectra.

1. Maneuver Loads

   a. Non-TFR Usage
   b. TFR Usage
   c. High Lift Usage

2. Gust Loads

   a. Non-TFR Usage
   b. TFR Usage
   c. High Lift Usage

The 4000 hour baseline spectrum load levels were combined into 1.0×10^6 in-lb increments and then the associated occurrences were divided by 4. The resulting two highest load levels were 14.0×10^6 in-lb and 13.0×10^6 in-lb with cumulative frequencies of 12 and 43, respectively. To obtain the one occurrence load levels for the 1000 hour spectrum, the 10 x 1000 hour spectrum and the 100 x 1000 hour spectrum a straight line extrapolation of the two highest load levels was used.

<u>4000 Hour Inspection Interval</u> - The 4000 hour interval spectrum is the repeated loads environment that would occur if the Phase I and II Training Syllabus were repeated approximately 54.6 times. This represents a typical airplane service life.

The cumulative frequency distribution is for a typical 4000 hour service life baseline spectrum. It includes the same loads spectra given above for the 25, 400, and 1000 hour distributions.

The load levels with their associated occurrences were sorted into 1.0×10^6 in-lb increments. The resulting two highest load levels were 15.0×10^6 in-lb and 14.0×10^6 in-lb with cumulative frequencies of 1 and 48, respectively. To obtain the one occurrence load levels for the 10 x 4000 hour spectrum and 100 x 4000 spectrum a straight line extrapolation of the two highest load levels was used.

IX.4 CONCLUSIONS AND RECOMMENDATIONS

The stated objective of the new damage tolerance criteria is to minimize service maintenance problems and to prevent the failure of "safety of flight" structure. The results of this study indicate that application of these new requirements to airframe structure is technically feasible, and should result in increased structural integrity. Delta cost estimates of fracture control impact to the F-111F baseline wing box (Paragraph IX.3.8) were conveyed separately to the Air Force.

There are some conclusions which can be summarized concerning the baseline studies conducted, and there are some points regarding the criteria which require additional clarification. These are discussed briefly below.

IX.4.1 Conclusions from Baseline Studies

The following conclusions are summarized based on the baseline sensitivity and trade studies

(1) Variation in 2024-T851 Al. K_{Ic} data (upper and lower versus middle bound) had a negligible effect (0 to 3%) on fracture design allowable stress levels determined for an 8000 hour period of unrepaired service usage. The effect for a 2000 hour period was somewhat greater (up to 10%). See Summary Table VIII. For reference, a 10% stress variation results in approximately 50 pounds delta weight for one wing. See Figure 40.

(2) Variation in 2024-T851 Al. da/dN data (upper and lower versus middle bound) impacted the fracture design stress levels by about the same amount (2 to 16%) as K_{Ic} variation for both the 2000 and 8000 hour periods of unrepaired service usage. See summary Table IX.

(3) Variations in both K_{Ic} and da/dN data resulted in significant life interval variation for a given constant stress level. See summary Tables X and XI.

(4) Variations in initial flaw size and in usage, of all the parameters investigated, indicated the greatest impact on both allowable stress and life. Mild usage provided as much as a 30% increase in allowable design stress.

(5) The impact of variation in residual strength load, determined from factored (1, 10 or 100) load exceedance curves for each inspection interval, on fracture design allowable stress level was most significant for the smaller inspection interval periods. See Table IX Higher allowable stresses were established for a given inspection interval using this new approach in lieu of assuming limit load, but this trend was very slight for an 8000 hour period of unrepaired service usage (non-inspectable).

(6) The variation in life intervals calculated for the current baseline lower skin stress level, using residual strength determined for load exceedance curve factors of 1, 10, and 100 was very slight. See Figure 91.

IX.4.2 Discussion of Revision D Proposed Damage Tolerance Requirements

The following comments are made on the proposed Air Force damage tolerance requirements (See Section IX.7).

(1) Fastener systems, particularly those utilizing interference fit, influence flaw propagation behavior. This should be recognized by the criteria. Perhaps smaller initial flaw sizes could be allowed in new structure if specified fastener installation controls are implemented. Analytically, the designer should be allowed the option, subject to Air Force Approval, of reflecting the fastener system in flaw growth models.

(2) Initial flaws at locations other than bolt holes are specified in terms of a/Q. The parameter Q is associated with the shape of part through flaws. Applying a/Q to "thick"plate structure involves assuming a shape since none is specified in the present criteria. Presumably, this would be interpreted as the "worst" shape.

Analytically, the worst shape is a long shallow flaw if the shape is assumed constant throughout propagation. However, testing experience indicates that such a shape will not remain constant. The problem is to define how the shape varies under spectrum loading.

The study in Section IX.5 assumed that a long shallow flaw ($a/2c = .1$) would grow in the depth dimension but surface length would remain fixed at its initial length. This assumption indicated no significant difference in allowable design stress level when compared to a semi-circular flaw shape ($a/2c = .5$) allowed to propagate as a constant semi-circular shape.

Another aspect of the assumed flaw shape question involves very "thin" sheet structure where initial part through flaws may essentially be through the thickness. Applying a/Q to this case indicates that a through the thickness "tear" may be more appropriate initial damage for thin sheet. One approach is to equate the flaw size parameter, $(K/\sigma)^2$, for an assumed part through shape (e.g., $a/2c = .5$) to the flaw size parameter for a through thickness flaw. For $a/Q = .03$, this indicates a through thickness flaw of initial length 0.06" in sheet thicknesses up to about 0.09 inches. Such a flaw length may be less than manufacturing NDI capability. Additional interpretation concerning initial flaws in thin sheet structure (e.g., the plies of a laminated lower wing skin) is needed in the criteria.

(3) The minimum assumed initial damage specified in the criteria following an in-service depot level inspection is much more severe than that specified for post manufacturing. The current alternatives are to design assuming non-inspectable structure, qualify NDI to detect smaller flaw sizes, or use proof test to establish smaller flaw sizes. Proof test is not a viable option except for some steel structures. The responsibility for performing depot inspections is usually beyond the control of the contractor. Therefore, the only real alternative for safety of flight is to design new structure as non-inspectable. This is particularly true for slow crack growth structure where the non-inspectable requirements are most severe. Of course, specific NDI requirements for critical structure (access, techniques, methods, etc.) could be developed for critical structure and negotiated with the AMMA's performing the inspections, but at increased cost to a program.

(4) The residual strength studies in paragraph IX.3.10, as applied to the F-111 baseline, indicate that the basic intent of the new criteria was not viable, i.e., the expected relief in the residual strength requirements for inspectable structure design as opposed to non-inspectable structure design, did not develop.

IX.4.3 NDI Demonstration Program Comments (reference paragraph 5.1.3 section (h) of MIL-STD-1530)

Cost studies performed during this program indicate that the type of specimen chosen for the NDI demonstration program greatly affects the cost. A study is recommended to provide statistical data on the significance of specimen complexity vs detectable defect size.

IX.5 ADDITIONAL STUDY OF ASSUMED FLAW SHAPES

The assumption that flaws propagate as a semi-circular flaw ($a/2c = .5$) is flet to be representative of flaw growth behavior exhibited by specimens tested using the baseline material and spectrum. However, additional studies involving another flaw shape assumption have been developed and compared with results developed for the semi-circular flaw shape.

This study was made for the part through flaw in the 0.611 inch thick wing skin. An initial flaw shape of $a/2c = .1$ was assumed. The flaw was allowed to propagate in depth, a, while holding the surface length, 2c, constant until the depth was equal to one half the surface length. At this point, any additional growth occurred as a semi-circular flaw. This analysis was accomplished by programming the crack growth computer procedure to vary both the shape ($a/2c$) and the backface correction factor (M_k) as the flaw growth progressed.

Sufficient crack growth analyses were performed with this procedure to allow development of design allowable curves for comparison with those in Figure 17 for mid-point fracture data. The resulting curve is given in Figure 92.

Entering the curves in Figure 92 and Figure 17 with an initial flaw size equivalent to $a/Q = .1$, the resulting comparison of allowable spectrum stresses is given below:

$a/2c = .1$, $Q = 1.10$, $a_o = .11''$, $2C_o = 1.10''$

$a/2c = .5$, $Q = 2.46$, $a_o = .246''$, $2C_o = .492''$

SHAPE	MAX. ALLOW. SPECTRUM STRESS, KSI			
	8000 Hr	4000 Hr	2000 Hr	800 Hr
a/2c = .1	19.0	21.7	25.7	--
a/2c = .5	19.7	21.9	24.4	26.7

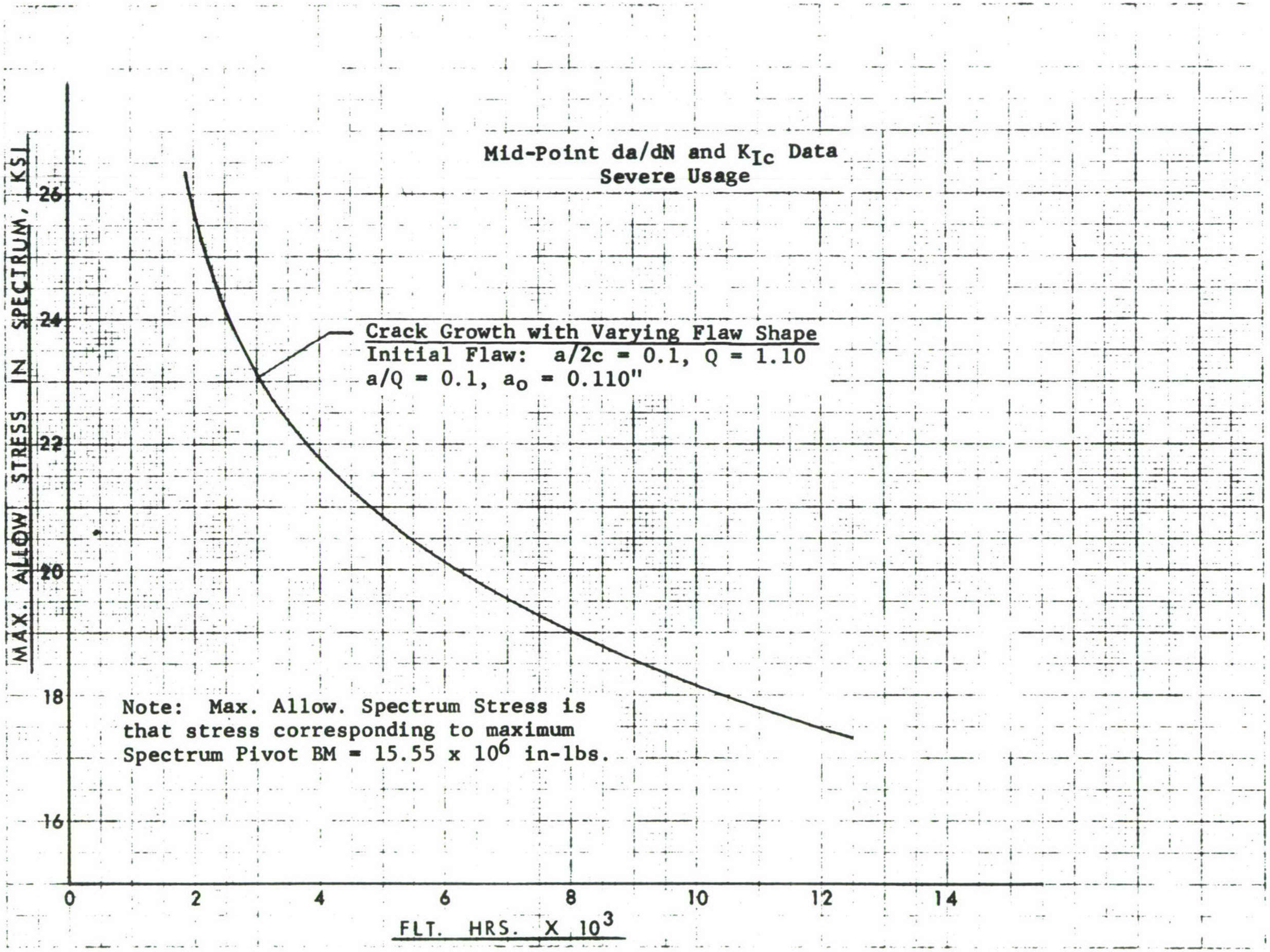


Figure 92 Flaw Shape Variation Allowable Curves for Baseline 2024-T851 Surface Flaws in 0.611 Skin

IX.6 BASELINE REDESIGN FOR DAMAGE TOLERANCE

A preliminary review of the analysis results generated during these studies was held at WPAFB with Air Force personnel on 15 and 16 February 1973. It was agreed to redesign the baseline wing to comply with the two lifetime (8000 hours) noninspectability requirements for slow crack growth structure as defined in Table I. It was also agreed that mid-point fracture data would be used for redesign.

Meeting these requirements was accomplished by adding additional material as necessary to hold lower wing skin stresses to the maximum allowable spectrum stress levels established using mid-point fracture data. The result is a delta weight increase. No material changes were made.

The weightvariation of the baseline wing box was presented in Figure 40 as a function of lower skin maximum allowable spectrum design stress level. This curve was used in conjunction with the design allowable curves in Figure 16 through 30 to determine the delta weight penalty resulting from maximum allowable stress levels dictated by variations in the damage tolerance requirements and/or analysis parameters.

The current maximum baseline lower surface stress level corresponding to maximum fatigue spectrum pivot bending moment is 25.2 ksi. The current weight of one baseline wing is therefore about 1550 pounds. The delta weight penalty is simply the difference between 1550 pounds and the new weight determined from Figure 40 for stresses meeting damage tolerance requirements. The minimum allowable spectrum stress in the lower wing skin (using mid-point data) is dictated by the data in Figure 26 for a through-the-thickness bolt hole flaw, i.e., 16.9 ksi for an assumed initial flaw 0.05". Table XXV summarizes the allowable stress for each of the flaw types evaluated in this study.

Entering Figure 40 with 16.9 ksi indicates that the wing will weigh about 1755 pounds. The maximum delta weight penalty is therefore (1755 -1550) = 205 pounds. This penalty is somewhat conservative because the damage tolerance criteria does not allow taking advantage of the generally acknowledged beneficial effect of the taper lok fastener system utilized on the baseline lower surface, i.e., open hole flaws are specified in the criteria with no additional policy recognizing the fastener system used.

Table XXV

F-111F BASELINE WING BOX
WEIGHT PENALTY SUMMARY
FOR 8000 HR. NONINSPECTABLE

Mid-Point da/dN Data, K_{IC} = 23 ksi $\sqrt{in}$
Residual Strength--Limit Load

Flaw Type	Max. Allow Spectrum Stress ksi	Weight Penalty per Wing, Lbs. Ref. Figure 3-40
Bolt Hole thru Flaw (Ref. Fig. 3-26)	16.9 (a_o = 0.05")	(1755 - 1550) = 205
Surf. Flaw, t = 0.611" (Ref. Fig. 3-17)	19.7 (a/Q = 0.1)	(1685 - 1550) = 135
Surf. Flaw, t = 1.30" (Ref. Fig. 3-20)	19.9 (a/Q = 0.1)	(1680 - 1550) = 130
Surf. Flaw, t = 0.25" (Ref. Fig. 3-23)	14.4 @ Spar Cap, or 17.4 @ Lwr Skin (a/Q = 0.1)	(1745 = 1550) = 195

The design allowable stress dictated by the spar cap part through flaw (t = .25") is shown in Table XXV as 14.4 ksi for 8000 hours non-inspectable. This stress level is determined from the curves in Figure 23. The spar cap stress level is less than the 16.9 ksi allowable for the 5/16" diameter lower surface bolt holes. However, the stresses at the spar caps are less than at the skin because the spar cap is located a shorter distance from the wing box neutral axis. When this is taken into account, the lower skin allowable stress required to hold the spar cap stress to 14.4 ksi is 17.4 ksi. Therefore, no material need be added to the spar caps, only the lower skin.

As discussed briefly above, the weight penalty of 205 pounds is considered conservative because the open hole flaw analysis does not reflect the retarding influence of properly installed taper lok fasteners. However, even when the bolt hole flaw is ignored, the lower skin stress allowable required to tolerate a spar cap surface flaw, results in a delta weight penalty of 195 pounds.

While the lack of a fastener policy is considered important, another conservatism contributing to the delta weight penalty of 205 pounds is the use of limit load in these studies as the residual strength requirement for flawed structure. The residual strength level is used to establish the critical flaw size, i.e., the flaw size at which unstable crack propagation occurs. As stated previously in paragraph IX.3.1.1, limit load is conservative when compared with the results shown in paragraph IX.3.10 where some residual strength sensitivity studies are presented. The effect of this conservatism on comparison of sensitivity and trade studies is not great, but the effect on redesign and weight penalty is more significant.

With this in mind, additional work has been performed to illustrate the baseline weight penalty associated with residual strength less than limit load.

There are three approaches to residual strength: (1) limit load (2) maximum spectrum load in a 4000 hour life determined from average load exceedance data, and (3) the new criteria definition involving factored load exceedance data.

The work in paragraph IX.3.10 is directed toward assessing the impact of residual strength as defined in the new criteria in Section IX.7. The new definition is that residual strength shall be the one occurrance load level determined from an average load-exceedance curve increased by a factor. The factor is 10 for fighter type aircraft. Using this definition indicates that the one occurrence load in one lifetime for the baseline is 15.6×10^6 in. lbs. pivot bending moment. This is less than the baseline limit bending moment of 19.52×10^6 in. lbs, and almost exactly identical to the maximum spectrum load level (15.55×10^6 in. lbs) previously quoted in this report. The factor of ten has no great impact due to the steep slope of the 4000 hour baseline exceedance curve at high load factor.

The design allowable curves in Figure 93 through 95 were developed using the above 15.6×10^6 in. lbs pivot bending moment for residual strength. The resulting maximum spectrum allowable stresses are 19.2 ksi for the bolt hole flaw (replaces 16.9 ksi) and 15.0 ksi for the spar cap surface flaw (replaces 14.4 ksi). The lower skin stress corresponding to the revised spar cap stress is 18.1 ksi. The revised delta weight penalties are shown in Table XXVI.

The actual structural changes necessary in the baseline wing structure to satisfy the damage tolerance criteria are shown in Drawing 610RW-000, reference Figure 96. These changes reflect a lower wing skin stress level of 16.9 ksi at maximum spectrum load. The actual delta weight required to achieve this stress level was calculated to be only 113.1 pounds rather than 205 pounds as determined from Figure 40. The possibility of this difference was previously discussed in Section IX.3, and is attributed to the fact that baseline skin stresses inboard of C.S.S. 140 are somewhat less than at that span station. Therefore, instead of a constant inboard increase in skin thickness, it was possible to use a variable thickness increase and still achieve the constant 16.9 ksi allowable stress.

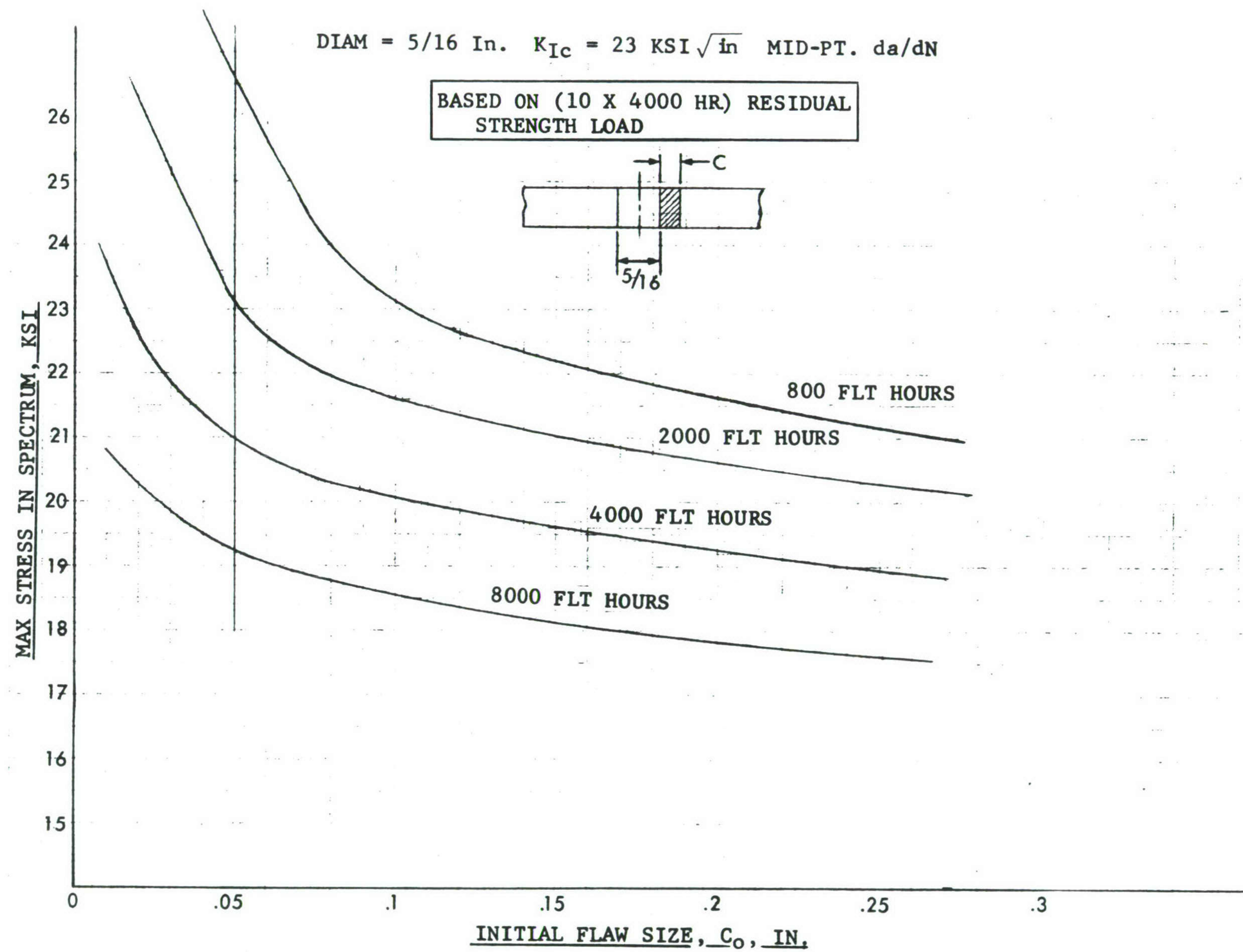


Figure 93 Allowable Curves for 2024-T851 Through Flaw at a Bolt Hole

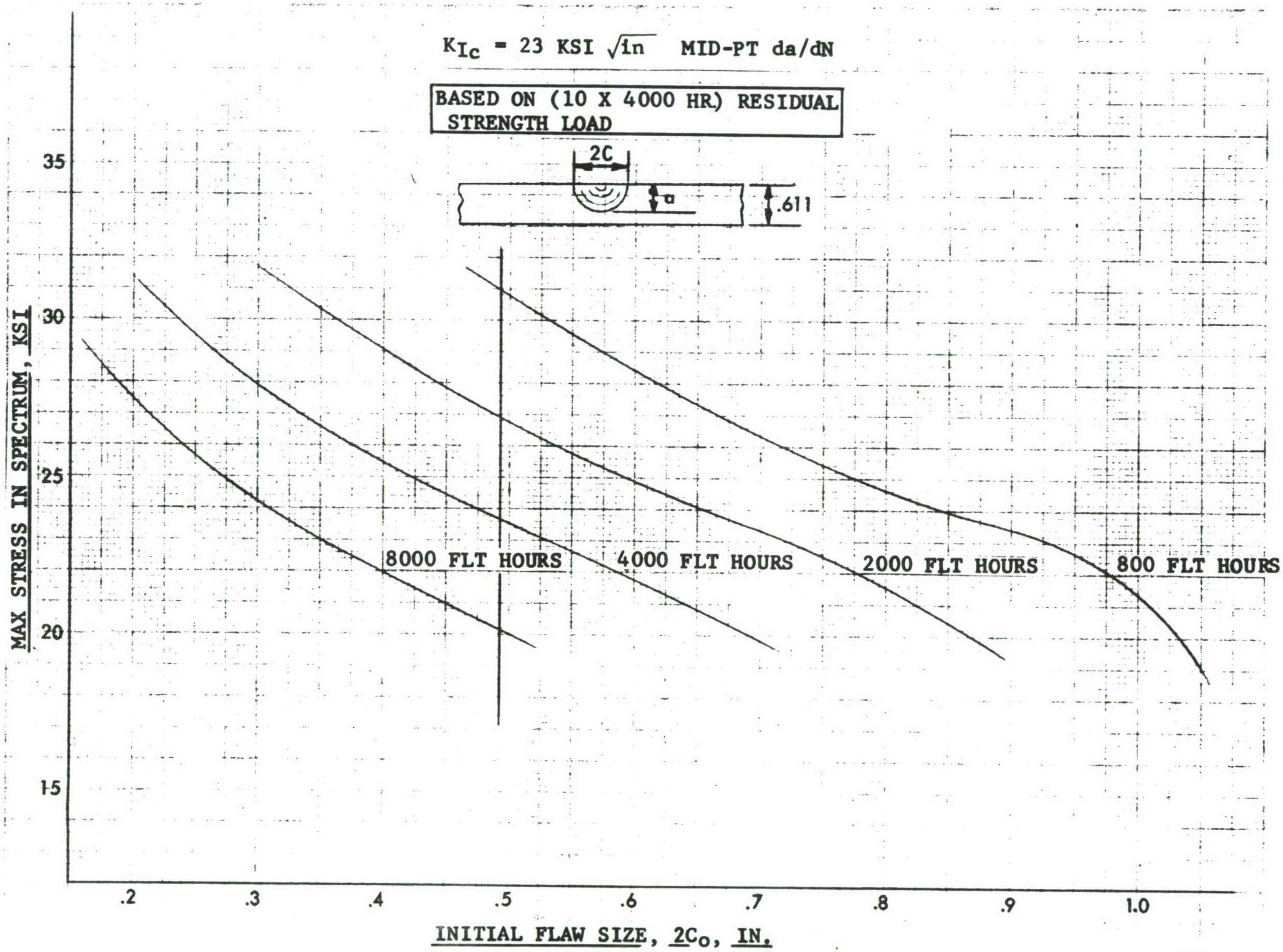


Figure 94 Allowable Curves for 2024-T851 Surface Flaw in .611 in. Skin

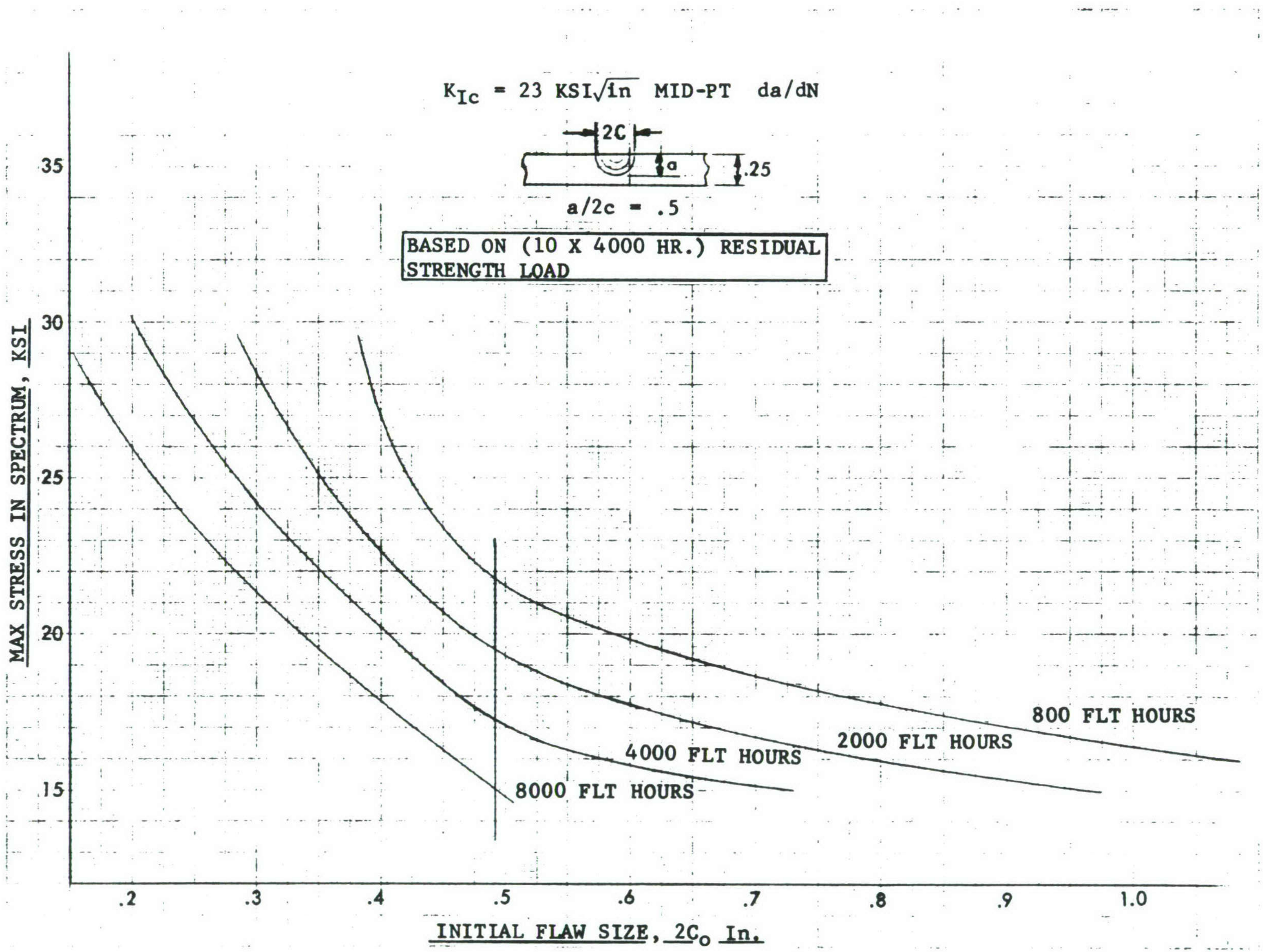


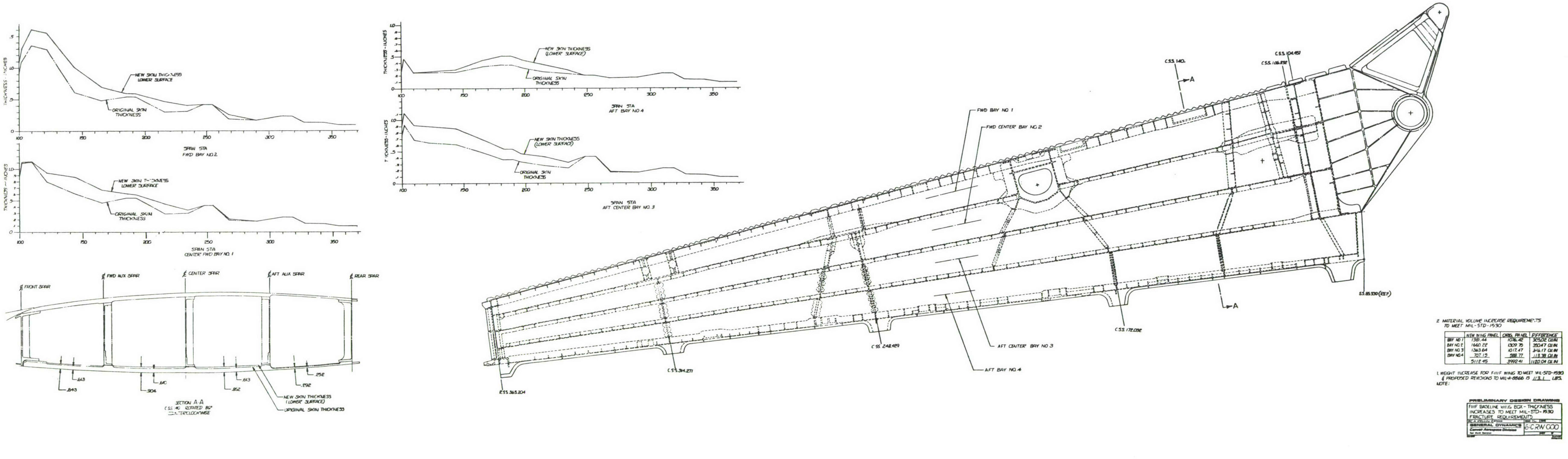
Figure 95 Allowable Curves for 2024-T851 Surface Flaw in .25 In Spar Cap

Table XXVI

F-111F BASELINE WING BOX
WEIGHT PENALTY SUMMARY
FOR 8000-HOUR NONINSPECTABLE

Mid-Point da/dN Data, $K_{IC} = 23 \text{ ksi} \sqrt{\text{in.}}$
Residual Strength--One Occurrence Load
in 10 x 4000 Flight Hours

Flaw Type	Max. Allow. Spectrum Stress ksi	Weight Penalty per Wing, Lbs. Ref. Figure 3-40
Bolt Hole thru Flaw (Ref. Fig. B-2)	19.2 (a_o = 0.05")	(1700 - 1550) = 150
Surf. Flaw, t = 0.611" (Ref. Fig. B-3)	20.2 (a/Q = 0.1)	(1675 - 1550) = 125
Surf. Flaw, t = 0.25" (Ref. Fig. B-4)	15.0 @ Spar Cap, or 18.1 @ Lwr Skin (a/Q = 0.1)	(1725 - 1550) = 175



2 MATERIAL VOLUME INCREASE REQUIREMENTS TO MEET MIL-STD-1530

	NEW WING PANEL	ORIG. PANEL	DIFFERENCE
BAY NO 1	1381.44	1076.42	305.02 CU.IN.
BAY NO. 2	1660.22	1309.75	350.47 CU.IN.
BAY NO. 3	1363.64	1017.47	346.17 CU.IN.
BAY NO. 4	707.15	588.77	113.38 CU.IN.
	5112.45	3992.41	1120.04 CU.IN.

1. WEIGHT INCREASE FOR F-111F WING TO MEET MIL-STD-1530 & PROPOSED REVISIONS TO MIL-A-8866 IS 113.1 LBS.
NOTE:

Figure 96 610RW000 F-111F Baseline Wing Box

IX.7 SUPPLEMENTAL DATA

The following General Dynamics Reports are presented in support of this report.

Supplement (A) - FZM-12-13249, Determination of Environmental Exposure of Critical F-111 Parts

Supplement (B) - F-111A D6Ac Steel Critical Part Temperature Study

Supplement (C) - MEA-301, A Summary of Nondestructive Inspection Performed On The F-111F Wing Box

Supplement (D) - M186 Standard, Serial Number Format, Traceability

Supplement (E) - Proposed Revision to MIL-A-8866 Dated 18 August 1972

Supplement (F) - MIL-STD-1530

Supplement (G) - Advanced Air Superiority Fighter Wing Structures Program - Follow-On Program Plan

SUPPLEMENT (A)

FZM-12-13249, DETERMINATION OF

ENVIRONMENTAL EXPOSURE OF CRITICAL

F-111 PARTS

FZM-12-13249
9 March 1971
Rev. 15 November 1972

DETERMINATION OF ENVIRONMENTAL EXPOSURE OF CRITICAL F-111 PARTS

(Title Unclassified)

Operations Research
GENERAL DYNAMICS
Convair Aerospace Division
Fort Worth Operation

ABSTRACT

From the flight loading test program established to investigate crack propagation in forged D6ac steel parts on the F-111, early results revealed that the crack growth rates in these parts are sensitive to certain chemical environments. These environments are moisture, humidity, fuel and lubricant.

This study has been conducted to determine the exposure of the critical parts to the above chemical environments which are expected to be encountered during F-111 operational usage. Basing data and the associated climatic data were combined to establish the chemical environments on the aircraft while it is in flight and on the ground. Flight training data for TAC training missions were analyzed to ascertain the lengths of these environmental exposures during the aircraft usage cycle.

The accessibility of the critical parts was established by determining the locations of the parts with respect to the access covers and drain holes on the aircraft. The above data on aircraft environments were then factored by the accessibility information to determine the flight and ground exposure times of the critical parts to the chemical environments.

NOTE: Utilization of the F-111 has changed somewhat since this analysis was originally performed (late 1970). The validity of the data in the light of these changes was briefly examined and the results of this examination are reported in Appendix A. It was concluded that the chemical environment now is actually less severe and therefore the data is conservative.

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1.0 INTRODUCTION

A study to investigate crack propagation in forged D6ac steel parts was established to determine if special IRAN (Inspect and Repair As Necessary) requirements are needed for these parts. This fracture mechanics study contains both analytical and test programs to determine the expected rate of crack propagation in the critical parts. Initial results from the test program indicated that exposure to chemical environments had a significant effect on the growth rate of cracks in these parts.

The present study was conducted to determine the exposure times of the critical parts to chemical environments which are expected to be encountered in F-111 operations. The environments of particular interest in this study are moisture, humidity, dry air, fuel and lubricant.

The study plan used to determine the environmental exposure of the critical parts is shown in Figure 1-1. The fifteen critical parts examined in this study are as follows:

1. Carry-Through-Box Outboard Bulkhead
2. Carry-Through-Box Aft Web
3. Carry-Through-Box Forward Web
4. Wing-Pivot-Fitting Upper Plate
5. Wing-Pivot-Fitting Lower Plate
6. Wing-Pivot-Fitting Forward Web
7. Wing-Pivot-Fitting Pivot Pin
8. Wing-Pivot-Fitting Shear Lug
9. Horizontal-Stabilizer Center Bulkhead
10. Horizontal-Stabilizer Upper Frame
11. Horizontal-Stabilizer Outboard Bulkhead

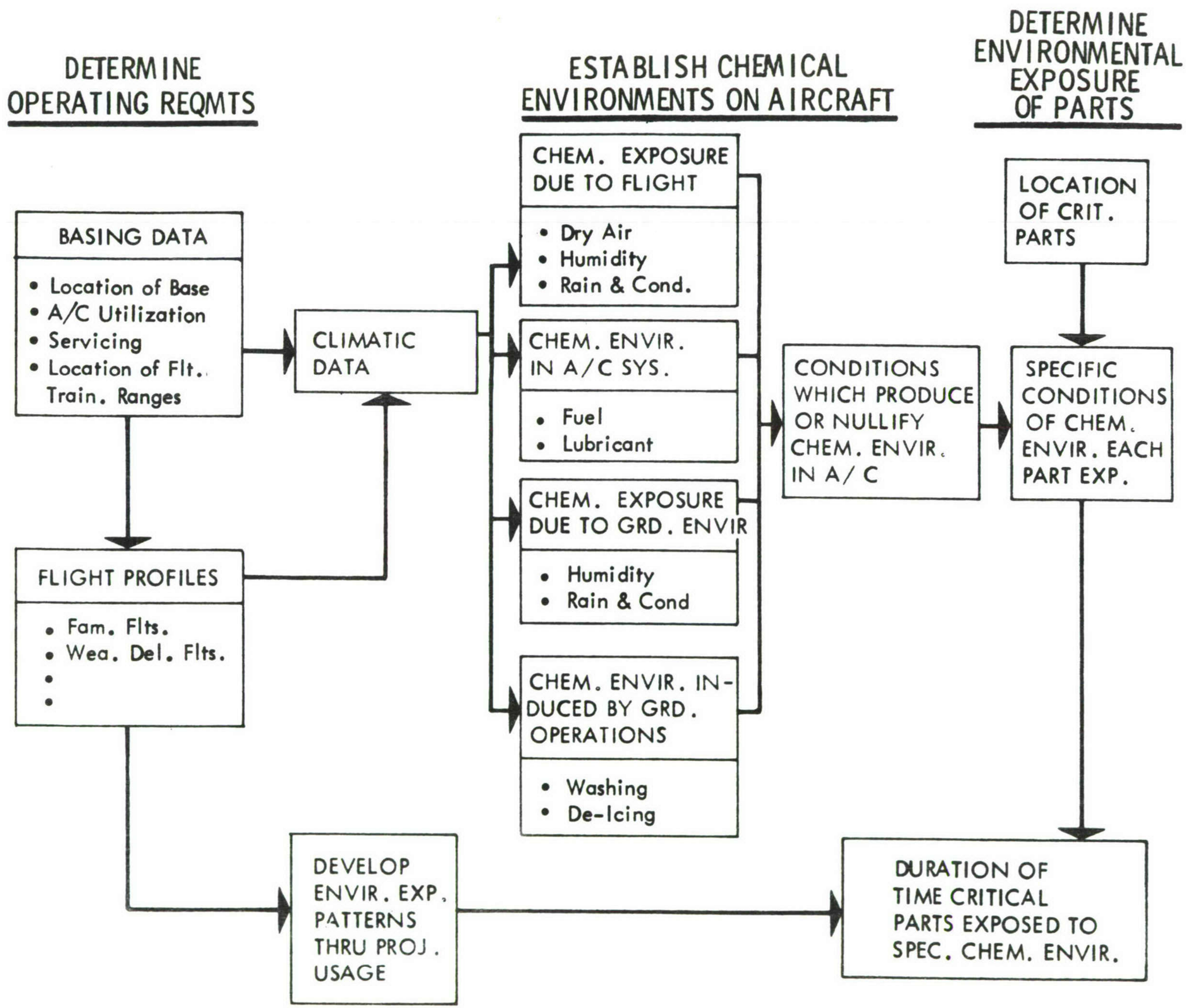


Figure 1-1 STUDY PLAN OF ENVIRONMENTAL EXPOSURE ANALYSES

12. Horizontal-Stabilizer Center Section

13. Upper Longeron

14. Station 496 Bulkhead Post

15. Rudder Torque Tube

As illustrated in the figure, the basing and flight training requirements must first be determined. These requirements are then combined with the attendant climatic data to determine the environmental exposure of the aircraft. Data on the location of critical parts on the aircraft must then be established; these data are used in factoring the aircraft environment to determine environmental exposure of the critical parts. The exposure times of the critical parts during the aircraft usage cycle are determined by analysis of the flight training mission requirements.

2.0 SUMMARY

The analyses to determine the exposure of the critical parts to chemical environments have been accomplished for the F-111s whose planned basing is at Nellis AFB, Upper Heyford, Cannon AFB and Homestead AFB. The results of these analyses are shown in Tables 2-1 through 2-4, respectively.*

An examination of the data in these tables reveals that they are different from the data in the Reference 1 memo. The data in this report are based on the length of the training cycle for the aircraft which is 1.4 times the length of the training cycle for the crews. The crew-training cycle was the basis of the exposure data in Reference 1; therefore, all the data have been scaled up by the 1.4 factor. *

The training cycle for the aircraft consists of 2.803 months, or 2046 hours. During the training cycle, the aircraft is in flight 98.2 hours while the remaining 1947.8 hours are spent on the ground.*

The part exposure to the environments encountered in flight can be summarized as follows:*

1. The average expected exposure of the parts to a moisture environment (rain and condensed humidity) is 2.15 hours. The Upper Heyford basing accounts for about 50 percent of this moisture exposure, and the basing at Homestead accounts for another 25 percent.

2. The average part exposure to humidity during flight is 47.1 hours. The relative humidity ranges from 35 percent for the western U. S. bases to 82 percent for Upper Heyford.

3. The flight exposure to fuel is 98.2 hours for the inner surfaces of all parts on the wing carry through box. For the inner surfaces of the parts on the wing pivot fitting, the fuel exposure is 17.2 hours with intermittent exposure to fuel during the other 81.0 hours of flight.

4. The wing pivot pin is exposed to the FMS-1071 lubricant during the total 98.2 hours of flight.

*Current operational data is somewhat different, see discussion in Appendix A.

Table 2-1 PART EXPOSURE DATA SHEET FOR F-111A BASED AT NELLIS AFB, NEVADA

EXPOSURE OF D6ac STEEL PARTS TO CHEMICAL ENVIRONMENT FOR F-111A BASED AT NELLIS AFB, NEV (Training Range at Edwards AFB Ranges)

ENVIRONMENT / PART	Ground Exposure (1947.8 hours/training cycle)							Flight Exposure (98.2 hours/training cycle)					
	HUMIDITY							HUMIDITY			DRY AIR		
	COND	0-25%	25-50%	50-75%	RAIN	FUEL	LUBR	COND	35%	RAIN		FUEL	LUBR
CTB OUTBD BHD 12B7313	-	805.5	1127.7	-	14.6	1947.8	-	1.02	48.2	0.04	48.9	98.2	-
CTB AFT WEB 12B7314	-	805.5	1127.7	-	14.6	1947.8	-	1.02	48.2	0.04	48.9	98.2	-
CTB FWD WEB 12B7315	-	805.5	1127.7	-	14.6	1947.8	-	1.02	48.2	0.04	48.9	98.2	-
WPF UPPER PLATE 12W475	-	805.5	1127.7	-	14.6	1947.8	-	1.02	48.2	0.04	48.9	98.2	-
WPF LOWER PLATE 12W476	-	805.5	1127.7	-	14.6	1947.8	-	1.02	48.2	0.04	48.9	17.2*	-
WPF FWD WEB 12W477	-	805.5	1127.7	-	14.6	1947.8	-	1.02	48.2	0.04	48.9	17.2*	-
WPF PIVOT PIN 12W415	-	-	-	-	-	-	1947.8	-	-	-	-	-	98.2
WPF SHEAR LUG 12W412	-	805.5	1127.7	-	14.6	-	-	1.02	48.2	0.04	48.9	-	-
770 CENTER BHD 12B10520	-	805.5	1127.7	-	14.6	-	-	1.02	48.2	0.04	48.9	-	-
770 UPPER FRAME 12B10523	-	805.5	1127.7	-	14.6	-	-	1.02	48.2	0.04	48.9	-	-
770 OUTBD BHD 12B10521	-	805.5	1127.7	-	14.6	-	-	1.02	48.2	0.04	48.9	-	-
HOR. TAIL BOX BEAM 12T9600	-	805.5	1127.7	-	14.6	-	-	1.02	48.2	0.04	48.9	-	-
UPPER LONGERON 12B7313	-	805.5	1127.7	-	14.6	-	-	1.02	48.2	0.04	48.9	-	-
496 BHD POST 12B2910	-	805.5	1127.7	-	14.6	-	-	1.02	48.2	0.04	48.9	-	-
RUDDER TORQ TUBE 12T406	-	805.5	1127.7	-	14.6	-	-	1.02	48.2	0.04	48.9	-	-

* Intermittent exposure for the remaining 81.0 flight hours.

Table 2-2 PART EXPOSURE DATA SHEET FOR F-111E BASED AT UPPER HEYFORD, ENGLAND

EXPOSURE OF D6ac STEEL PARTS TO CHEMICAL ENVIRONMENTS FOR F-111E BASED AT UPPER HEYFORD, ENGLAND (Training Range at English Ranges)

ENVIRONMENT / PART	Ground Exposure (1947.8 hours/training cycle)							Flight Exposure (98.2 hours/training cycle)					
	HUMIDITY							HUMIDITY					
	COND	25-50%	50-75%	75-100%	RAIN	FUEL	LUBR	COND	75%	RAIN	DRY AIR	FUEL	LUBR
CTB OUTBD BHD 12B7313	383.7	-	686.4	663.9	213.8	1947.8	-	3.68	44.8	0.71	48.9	98.2	-
CTB AFT WEB 12B7314	383.7	-	686.4	663.9	213.8	1947.8	-	3.68	44.8	0.71	48.9	98.2	-
CTB FWD WEB 12B7315	383.7	-	686.4	663.9	213.8	1947.8	-	3.68	44.8	0.71	48.9	98.2	-
WPF UPPER PLATE 12W475	383.7	-	686.4	663.9	213.8	1947.8	-	3.68	44.8	0.71	48.9	17.2*	-
WPF LOWER PLATE 12W476	383.7	-	686.4	663.9	213.8	1947.8	-	3.68	44.8	0.71	48.9	17.2*	-
WPF FWD WEB 12W477	383.7	-	686.4	663.9	213.8	1947.8	-	3.68	44.8	0.71	48.9	17.2*	-
WPF PIVOT PIN 12W415	-	-	-	-	-	-	1947.8	-	-	-	-	-	98.2
WPF SHEAR LUG 12W412	383.7	-	686.4	663.9	213.8	-	-	3.68	44.8	0.71	48.9	-	-
770 CENTER BHD 12B10520	383.7	-	686.4	663.9	213.8	-	-	3.68	44.8	0.71	48.9	-	-
770 UPPER FRAME 12B10523	383.7	-	686.4	663.9	213.8	-	-	3.68	44.8	0.71	48.9	-	-
770 OUTBD BHD 12B10521	383.7	-	686.4	663.9	213.8	-	-	3.68	44.8	0.71	48.9	-	-
HOR. TAIL BOX BEAM 12T9600	383.7	-	686.4	663.9	213.8	-	-	3.68	44.8	0.71	48.9	-	-
UPPER LONGERON 12B7313	383.7	-	686.4	663.9	213.8	-	-	3.68	44.8	0.71	48.9	-	-
496 BHD POST 12B2910	383.7	-	686.4	663.9	213.8	-	-	3.68	44.8	0.71	48.9	-	-
RUDDER TORQ TUBE 12T406	383.7	-	686.4	663.9	213.8	-	-	3.68	44.8	0.71	48.9	-	-

* Intermittent exposure for the remaining 81.0 flight hours

Table 2-3 PART EXPOSURE DATA SHEET FOR F-111D BASED AT CANNON AFB, NEW MEXICO

EXPOSURE OF D6ac STEEL PARTS TO CHEMICAL ENVIRONMENTS FOR F-111D BASED AT CANNON AFB, N. M. (Training Range at Edwards AFB Ranges)

ENVIRONMENT	Ground Exposure (1947.8 hours/training cycle)							Flight Exposure (98.2 hours/training cycle)					
	HUMIDITY							HUMIDITY					
PART	COND	0-25%	25-50%	50-75%	RAIN	FUEL	LUBR	COND	35%	RAIN	DRY AIR	FUEL	LUBR
CTB OUTBD BHD 12B7313	18.8	-	1277.7	620.2	31.1	1947.8	-	1.02	48.2	0.04	48.9	98.2	-
CTB AFT WEB 12B7314	18.8	-	1277.7	620.2	31.1	1947.8	-	1.02	48.2	0.04	48.9	98.2	-
CTB FWD WEB 12B7315	18.8	-	1277.7	620.2	31.1	1947.8	-	1.02	48.2	0.04	48.9	98.2	-
WPF UPPER PLATE 12W475	18.8	-	1277.7	620.2	31.1	1947.8	-	1.02	48.2	0.04	48.9	17.2*	-
WPF LOWER PLATE 12W476	18.8	-	1277.7	620.2	31.1	1947.8	-	1.02	48.2	0.04	48.9	17.2*	-
WPF FWD WEB 12W477	18.8	-	1277.7	620.2	31.1	1947.8	-	1.02	48.2	0.04	48.9	17.2*	-
WPF PIVOT PIN 12W415	-	-	-	-	-	-	1947.8	-	-	-	-	-	98.2
WPF SHEAR LUG 12W412	18.8	-	1277.7	620.2	31.1	-	-	1.02	48.2	0.04	48.9	-	-
770 CENTER BHD 12B10520	18.8	-	1277.7	620.2	31.1	-	-	1.02	48.2	0.04	48.9	-	-
770 UPPER FRAME 12B10523	18.8	-	1277.7	620.2	31.1	-	-	1.02	48.2	0.04	48.9	-	-
770 OUTBD BHD 12B10521	18.8	-	1277.7	620.2	31.1	-	-	1.02	48.2	0.04	48.9	-	-
HOR. TAIL BOX BEAM 12T9600	18.8	-	1277.7	620.2	31.1	-	-	1.02	48.2	0.04	48.9	-	-
UPPER LONGERON 12B7313	18.8	-	1277.7	620.2	31.1	-	-	1.02	48.2	0.04	48.9	-	-
496 BHD POST 12B2910	18.8	-	1277.7	620.2	31.1	-	-	1.02	48.2	0.04	48.9	-	-
RUDDER TORQ TUBE 12T406	18.8	-	1277.7	620.2	31.1	-	-	1.02	48.2	0.04	48.9	-	-

* Intermittent exposure for the remaining 81.0 flight hours.

Table 2-4 PART EXPOSURE DATA SHEET FOR F-111F BASED AT HOMESTEAD AFB, FLORIDA

EXPOSURE OF D6ac STEEL PARTS TO CHEMICAL ENVIRONMENTS FOR F-111F BASED AT HOMESTEAD AFB, FLA (Training Range at Eglin AFB Ranges)

	Ground Exposure (1947.8 hours/training cycle)							Flight Exposure (98.2 hours/training cycle)					
ENVIRONMENT	HUMIDITY							HUMIDITY					
PART	COND	25-50%	50-75%	75-100%	RAIN	FUEL	LUBR	COND	75%	RAIN	DRY AIR	FUEL	LUBR
CTB OUTBD BHD 12B7313	170.5	-	830.6	812.4	134.3	1947.8	-	1.61	47.2	0.46	48.9	98.2	-
CTB AFT WEB 12B7314	170.5	-	830.6	812.4	134.3	1947.8	-	1.61	47.2	0.46	48.9	98.2	-
CTB FWD WEB 12B7315	170.5	-	830.6	812.4	134.3	1947.8	-	1.61	47.2	0.46	48.9	98.2	-
WPF UPPER PLATE 12W475	170.5	-	830.6	812.4	134.3	1947.8	-	1.61	47.2	0.46	48.9	17.2*	-
WPF LOWER PLATE 12W476	170.5	-	830.6	812.4	134.3	1947.8	-	1.61	47.2	0.46	48.9	17.2*	-
WPF FWD WEB 12W477	170.5	-	830.6	812.4	134.3	1947.8	-	1.61	47.2	0.46	48.9	17.2*	-
WPF PIVOT PIN 12W415	-	-	-	-	-	-	1947.8	-	-	-	-	-	98.2
WPF SHEAR LUG 12W412	170.5	-	830.6	812.4	134.3	-	-	1.61	47.2	0.46	48.9	-	-
770 CENTER BHD 12B10520	170.5	-	830.6	812.4	134.3	-	-	1.61	47.2	0.46	48.9	-	-
770 UPPER FRAME 12B10523	170.5	-	830.6	812.4	134.3	-	-	1.61	47.2	0.46	48.9	-	-
770 OUTBD BHD 12B10521	170.5	-	830.6	812.4	134.3	-	-	1.61	47.2	0.46	48.9	-	-
HOR. TAIL BOX BEAM 12T9600	170.5	-	830.6	812.4	134.3	-	-	1.61	47.2	0.46	48.9	-	-
UPPER LONGERON 12B7313	170.5	-	830.6	812.4	134.3	-	-	1.61	47.2	0.46	48.9	-	-
496 BHD POST 12B2910	170.5	-	830.6	812.4	134.3	-	-	1.61	47.2	0.46	48.9	-	-
RUDDER TORQ TUBE 12T406	170.5	-	830.6	812.4	134.3	-	-	1.61	47.2	0.46	48.9	-	-

* Intermittent exposure for the remaining 81.0 flight hours

The environmental exposure of parts while on the ground can be summarized as follows:*

1. The average ground exposure of the parts to moisture (rain and condensed humidity) is 241.7 hours. The aircraft based at Upper Heyford receive about 60% of this exposure, and the Homestead basing accounts for another 30 percent.

2. The average part exposure to relative humidity above 50 percent is 903.4 hours while the exposure to relative humidity below 50 percent is 802.7 hours. The exposure to the higher humidity range is attributable almost entirely to the basing at Upper Heyford and Homestead, and the exposure to the lower humidity range results entirely from basing at Nellis and Cannon.

3. The fuel exposure while the aircraft is on the ground is 1947.8 hours for the inner surfaces of all parts on the wing carry through box and the wing pivot fitting.

4. The exposure of the wing pivot pin to the lubricant is 1947.8 hours.

Exposures to environments other than those of primary interest in the study have been noted. These environments include: (1) smoke, sulphur dioxide and salt concentrations present in the air at Upper Heyford and Homestead; (2) hydraulic fluid for those parts in proximity to the actuators in the hydraulic system, and (3) cleaning compounds and solvents induced on the aircraft by ground operations.

*See Appendix A.

3.0 BASING AND FLIGHT TRAINING DATA *

In order to determine the expected usage of the F-111s as they are phased into the Air Force inventory, it was necessary to establish the basing and associated flight training data for the aircraft. These data were utilized to ascertain the climatic data required, the aircraft usage cycle and information on the flight training profiles of the aircraft. Although the information presently available can not project every basing and flight environment which will be encountered during the life of the aircraft, it does represent a wide range of environments resulting from present employment/deployment planning.

3.1 Data on Basing Locations. The basing locations of the F-111s entering the Air Force inventory, according to the present planning in Reference 2, are shown in Table 3-1. The locations of the flight training ranges shown in the table

TABLE 3-1 Locations of F-111 Basing and Flight Training Ranges

Aircraft Model	Basing Location	Flight Training Range Location
F-111A	Nellis AFB, Nevada	Edwards AFB Ranges
F-111D	Cannon AFB, New Mexico	Edwards AFB Ranges
F-111E	Upper Heyford, England	Isle of Man, English Low-Level Link Routes and RBS Sites
F-111F	Homestead AFB, Florida	Eglin AFB Ranges

in conjunction with the bases located in the U.S. are predicated on discussions with Convair Aerospace Division personnel associated with the planning of the training programs.

*See Appendix A

The locations of the flight training ranges for Upper Heyford were obtained from the Reference 3 letter. This letter stated that Phase III training flights over French ranges are being considered; however, there is no firm information that these ranges would be available to U.S. aircraft. The English training ranges currently being used, therefore, were listed in Table 3-1.

3.2 Aircraft Utilization. The utilization of the F-111s during the Phase III training program is used to determine the length of the training cycle in terms of the aircraft. In addition, the relative exposure to ground and flight environments during this period can be determined.

An operational F-111 wing is scheduled to have 72 aircraft which are to be flown by 90 combat crews and 10 staff crews. This results in a crew-to-aircraft ratio of 1.4. Since each crew is required to fly 25 hours per month, the aircraft utilization will be 35 hours per aircraft per month.

In accordance with Reference 4, the training cycle for each crew during Phase III training consists of 70.1 flight hours. With a crew ratio of 1.4, the length of the training cycle in terms of the aircraft is 98.2 hours. At 35 hours per month, the training cycle for the aircraft lasts 2.803 months, 85.25 days or 2046.0 hours. Since there are 98.2 flight hours per training cycle for the aircraft, the aircraft is on the ground the remaining time, or 1947.8 hours.

3.3 Flight Training Profiles. The Phase III (continuation) Flight Training program was derived in Reference 4 and represents a compendium of Air Force continuation training requirements. This crew training consists of 22 flights with a total flight time of 70.1 hours. The average flight length, therefore, is 3.19 hours in this training program.

The training missions were segmented into flight times above and below 5000-foot altitude as shown in Table 3-2. In these missions, the level-flight altitudes below 5000 feet are nominally between 200 and 1000 feet, and the level-flight altitudes above 5000 feet are from 18,000 to 25,000 feet. The remaining flight time is consumed by ascending to or descending from these altitudes.

The flight altitudes were analyzed in the above manner to aid in establishing the environmental exposure of the aircraft during flight. It was assumed the flight altitudes below 5000 feet would occur when the aircraft is flying at low altitude over the flight training ranges, and thus exposed

TABLE 3-2 Analysis of Flight Altitudes During Phase III Flight Training

Mission	Flight Time Below 5000 Ft - Minutes		Flight Time Above 5000 Ft - Minutes
	200-1000 Ft	Ascent & Descent < 5000 Ft	
TCT-1	50	23	56
TCT-2	2	11	107
TCT-3	37	36	111
TCT-4	65	32	81
TCT-5	91	16	127
TCT-6	84	26	84
TCT-7	84	26	84
TCT-8	84	26	84
TCT-9	84	26	84
TCT-10	84	26	84
TCT-11	84	26	84
TCT-12	105	24	121
TCT-13	85	25	39
TCT-14	29	9	117
TCT-15	--	8	335
TCT-16	100	9	98
TCT-17	140	6	--
CPM-1	81	8	91
CPM-2	81	8	91
CPM-3	100	9	98
CPM-4	102	10	122
CPM-5	141	12	--
TOTAL	2115		2098

to the rain and humid air in that area. The air is relatively dry above 5000 feet, and any clouds or rain at the 18,000-to-25,000-foot level is normally avoided without difficulty.

Considering the crew-to-aircraft ratio, there are 30.8 aircraft flights per training cycle. As shown in Table 3-2, the time above 5000 feet is 1.59 hours; therefore, the flight time per training cycle when the dry air environment is encountered is 48.9 hours. The time below 5000 feet is 1.6 hours, and the flight time per training cycle when rain and humid air are encountered is 49.3 hours.

4.0 CLIMATIC DATA *

The climatic data for the different bases where the F-111s will be operated has been assembled for use in establishing the chemical environments to which the aircraft will be exposed. The data for ground conditions which were obtained from Reference 5 are shown for Nellis AFB, Upper Heyford, Cannon AFB and Homestead AFB in Tables 4-1 through 4-4, respectively.

Additional data on ground conditions at Upper Heyford, England were received in Reference 6. This information, which was obtained from sites monitoring air pollution levels, revealed that yearly average concentrations of 31 micrograms per cubic centimeter for smoke and 55 micrograms per cubic centimeter for sulphur dioxide had been measured for the period 1 April 1969 to 31 March 1970.

The climatic data for the conditions encountered during flight were estimated for the areas of the Eglin and Edwards test ranges from data contained in Reference 7. The data for flight conditions on the English flight ranges were assumed to be the same as the climatic data for Upper Heyford since data relating specifically to these ranges were not available. The climatic data used for the flight training ranges are shown in Tables 4-5 through 4-7.

*See Appendix A

Table 4-1 CLIMATIC DATA FOR NELLIS AFB, NEVADA

PARAMETER	JAN	FEB	MAR	APR	MAY	JUN	JUL	AUG	SEPT	OCT	NOV	DEC	ANNUAL
Mean Monthly Temp (^{o}F)	45	50	56	66	74	83	90	88	81	69	55	47	
Relative Humidity (%)	47	40	31	26	21	18	21	25	24	30	40	47	
Dew Point Temp (^{o}F)	23	24	22	26	29	33	42	45	37	32	27	24	
Days of Precipitation	2	2	2	2	1	-	2	2	1	2	2	2	20
Mean Monthly Precip (in)	0.4	0.3	0.2	0.4	0.1	-	0.3	0.5	0.3	0.2	0.4	0.3	3.3

Table 4-2 CLIMATIC DATA FOR UPPER HEYFORD, ENGLAND

PARAMETER	JAN	FEB	MAR	APR	MAY	JUN	JUL	AUG	SEPT	OCT	NOV	DEC	ANNUAL
Mean Monthly Temp (^{o}F)	37	38	43	48	54	58	61	61	57	51	44	41	
Relative Humidity (%)	89	87	82	78	78	74	76	79	82	86	90	90	
Dew Point Temp (^{o}F)	35	36	38	41	47	49	53	55	52	47	42	40	
Days of Precipitation	22	19	18	18	16	18	18	19	18	20	21	22	229
Mean Monthly Precip (in)	2.4	1.3	1.5	1.5	1.9	2.0	1.7	2.1	1.7	2.6	2.8	2.3	23.8

Table 4-3 CLIMATIC DATA FOR CANNON AFB, NEW MEXICO

PARAMETER	JAN	FEB	MAR	AP	MAY	JUN	JUL	AUG	SEPT	OCT	NOV	DEC	ANNUAL
Mean Monthly Temp (^{o}F)	38	42	47	57	66	75	78	77	70	59	47	39	
Relative Humidity (%)	55	53	44	41	45	48	52	54	56	53	53	57	
Dew Point Temp (^{o}F)	20	22	22	29	39	50	56	56	50	38	27	22	
Days of Precipitation	3	3	3	3	5	6	8	6	6	4	3	3	53
Mean Monthly Precip (in)	0.5	0.4	0.4	0.6	1.3	2.0	2.9	1.0	1.4	1.5	0.4	0.5	13.5

Table 4-4 CLIMATIC DATA FOR HOMESTEAD AFB, FLORIDA

PARAMETER	JAN	FEB	MAR	APR	MAY	JUN	JUL	AUG	SEPT	OCT	NOV	DEC	ANNUAL
Mean Monthly Temp (°F)	66	68	71	74	77	80	82	83	81	77	73	67	
Relative Humidity (%)	75	73	71	70	74	77	77	77	78	76	74	73	
Dew Point Temp (°F)	58	59	61	64	68	72	73	74	73	69	64	58	
Days of Precipitation	7	5	6	4	9	13	14	15	16	13	4	4	114
Mean Monthly Precip (in)	2.2	1.9	2.7	1.9	6.2	7.9	6.7	6.0	8.9	8.1	2.6	1.6	56.7

Table 4-5 CLIMATIC DATA FOR EDWARDS AFB FLIGHT RANGES

PARAMETER	JAN	FEB	MAR	APR	MAY	JUN	JUL	AUG	SEPT	OCT	NOV	DEC	ANNUAL
Mean Monthly Temp (^{O}F)	51	56	62	72	80	89	95	94	85	75	61	53	
Relative Humidity (%)	51	44	35	30	24	22	25	29	29	34	44	51	
Dew Point Temp (^{O}F)	34	35	34	39	40	45	52	55	48	45	38	36	
Days of Precipitation	3	3	3	2	2	1	2	2	1	2	2	3	26
Mean Monthly Precip (in)	0.9	1.0	0.7	0.5	0.2	0.1	0.2	0.4	0.3	0.4	0.6	1.0	6.3

Table 4-6 CLIMATIC DATA FOR ENGLISH FLIGHT RANGES

PARAMETER	JAN	FEB	MAR	APR	MAY	JUN	JUL	AUG	SEPT	OCT	NOV	DEC	ANNUAL
Mean Monthly Temp (^{o}F)	37	38	43	48	54	58	61	61	57	51	44	41	
Relative Humidity (%)	89	87	82	78	78	74	76	79	82	86	90	90	
Dew Point Temp (^{o}F)	35	36	38	41	47	50	52	55	52	46	42	40	
Days of Precipitation	22	19	18	18	16	18	18	19	18	20	21	22	229
Mean Monthly Precip (in)	2.4	1.3	1.5	1.5	1.9	2.0	1.7	2.1	1.7	2.6	2.8	2.3	23.8

Table 4-7 CLIMATIC DATA FOR EGLIN AFB FLIGHT RANGES

PARAMETER	JAN	FEB	MAR	APR	MAY	JUN	JUL	AUG	SEPT	OCT	NOV	DEC	ANNUAL
Mean Monthly Temp (^{o}F)	54	56	61	68	75	80	81	81	78	70	57	54	
Relative Humidity (%)	75	75	73	75	70	77	80	79	77	73	70	75	
Dew Point Temp (^{o}F)	46	48	52	59	64	73	76	75	69	61	45	46	
Days of Precipitation	9	9	10	7	7	11	15	13	10	6	6	9	112
Mean Monthly Precip (in)	3.9	4.0	5.5	4.8	4.0	5.2	8.1	7.3	5.4	2.2	3.0	4.2	57.6

5.0 CHEMICAL ENVIRONMENT ON THE AIRCRAFT

The chemical environments on the aircraft are the combined results of the ground environment, the environment encountered in flight, the environment inherent on the aircraft and the environment induced by ground operations. The environments of particular interest in this study are dry air, humidity, water and fuel; however, other environments which have been found in the development of the analysis are discussed.

5.1 Ground Environment - The environments prevalent when the aircraft are on the ground are rain and humidity. The expected exposure of the aircraft to rainfall during a training cycle can be determined by the equation

$$T_R = (N_R)(H_{R/R})$$

where N_R = expected days of rain during a training cycle

$H_{R/R}$ = expected hours of rain, given that it rains

The expected days of rain during a training cycle can be expressed as

$$N_R = (P_R)(N)$$

where P_R = probability of rain on a given day

N = number of days in the training cycle

and, $P_R = \dfrac{\text{Number of days of rain per year}}{\text{Number of days per year}}$

The expected hours of rain, given that it rains, can be expressed as

$$H_{R/R} = (A_{R/R})/(R)$$

where $A_{R/R}$ = average inches of rain, given that it rains

R = rainfall rate in inches per hour

$$\text{and, } A_{R/R} = \frac{\text{Average annual rainfall}}{\text{Number of days of rain per year}}$$

The data for Homestead AFB is utilized in an example which illustrates the computation of the expected exposure of the aircraft to rain. The climatic data for this base includes 115 days of rain annually and the average annual rainfall is 56.7 inches; after reviewing the rainfall data, the average rainfall rate was estimated to be 0.1 inches per hour. As determined in Section 3.0, there are 85.25 days in a training cycle.

From the above data, the expected exposure of the aircraft to rainfall during the training cycle is computed as follows:

$$P_R = 115/365 = .315$$

$$N_R = (.315)(85.25) = 26.85 \text{ days}$$

$$H_{R/R} = (56.7/115.0)(1.0/0.1) = 5.0 \text{ hours}$$

$$T_R = (26.85)(5.0) = 134.3 \text{ hours per training cycle}$$

The expected rainfall exposure for the aircraft was computed in a similar manner for the other bases except that the average rainfall rate for Upper Heyford was estimated to be 0.025 inches per hour.

The expected humidity exposure of the aircraft per training cycle was determined by estimating the average daily relative humidity cycle using References 5 and 7. This relative humidity cycle for Homestead AFB is shown in Figure 5-1.

The humidity exposure of the aircraft was measured in terms of the time the relative humidity was in designated ranges (i.e., 0-25 percent, 25-50 percent, 50-75 percent and 75-100 percent). As can be seen in Figure 5-1, the relative humidity is never below 50 percent; it is in the 50-75 percent range 11 hours per day or 45.8 percent of the time, and in the 75-100 percent range 13 hours per day or 54.2 percent of the time.

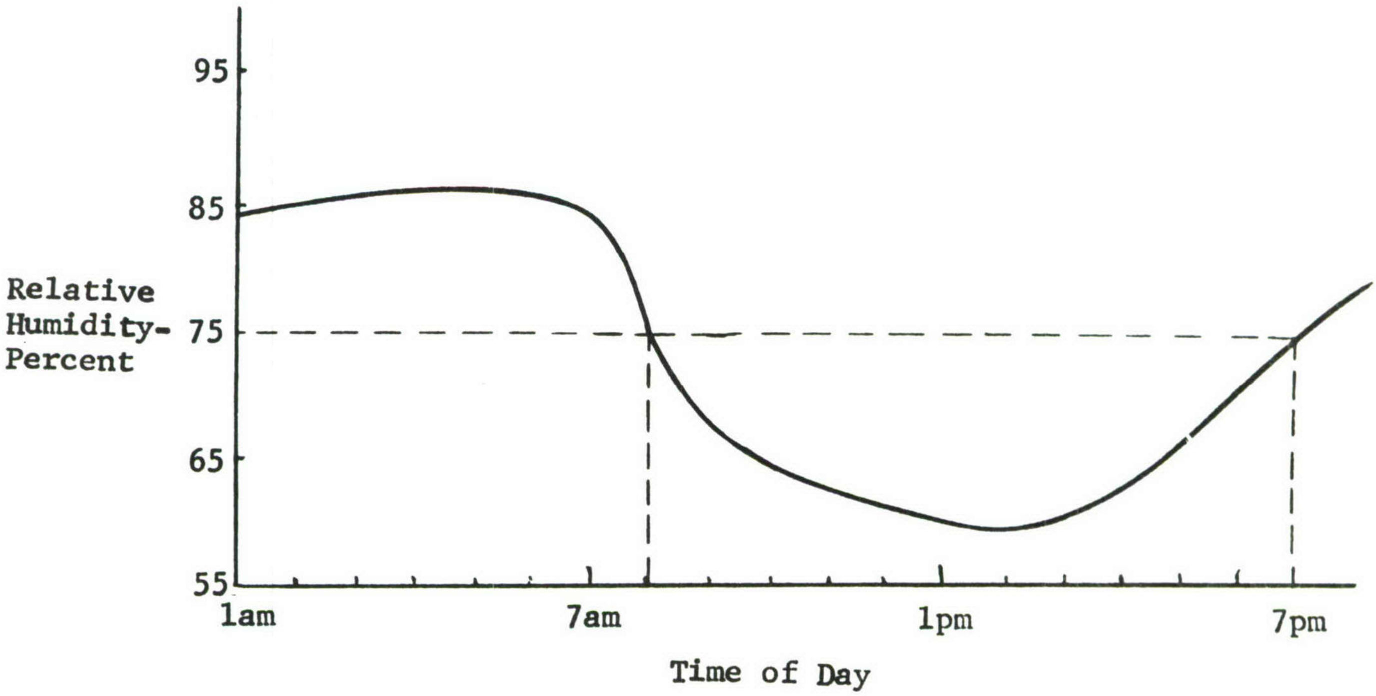


Figure 5-1 Estimated Daily Humidity Cycle at Homestead AFB

Another environment which must be considered in conjunction with the higher relative humidities is the condensation of airborne moisture on the aircraft. This condensation can be expected to occur during the pre-dawn hours when the aircraft surfaces temperatures drop (due to cumulative radiation heat losses) below the dewpoint temperature (a function of the relative humidity). The exposure to condensation on the external surfaces of the aircraft was not determined since it was assumed the critical parts are not subjected to it; however, humidity condensation on the critical parts can occur and the part exposure to this environment is discussed in more detail in the next section.

The aircraft is also exposed to other environments while on the ground. The data in Reference 6 shows smoke and sulphur dioxide concentrations of 0.88 and 1.56 per cubic foot, respectively, at the Upper Heyford base. Also, there are salt concentrations of unknown levels at Homestead AFB due to the proximity of the ocean.

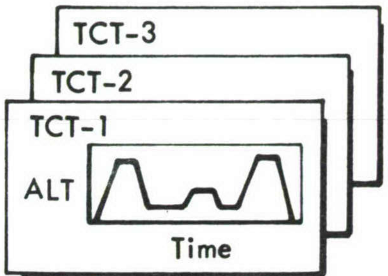
The above concentrations of contaminants in the air combined with the high humidity levels at these bases could result in serious corrosion problems on aircraft surfaces. These contaminants would tend to build up relatively fast on the external surfaces; however, rain exposure and washing of the aircraft as required in ground operations should control the corrosive effects of these concentrations.

5.2 Flight Environment. The environments encountered by the aircraft during flight are dry air, rain and humidity. As discussed in Section 3.3, the flight time per training cycle when the aircraft encounters the dry air environment is 48.9 hours. The rain and humidity environment are encountered 49.3 hours per training cycle.

An example of the methodology employed to determine the aircraft rain exposure expected during training flights is shown in Figure 5-2. As can be seen in the figure, the climatic data used in this example are those for the Eglin AFB flight ranges which were tabulated in Section 4.0; the aircraft utilization and mission profile data used in the example were developed in Section 3.0.

EXPOSURE TO RAIN DURING FLIGHT

FLIGHT PROFILES



ANALYSIS OF FLIGHT PROFILES

	TIME AT ALTITUDES - (MIN) 200'	500'	1000'	>5000'	ASC/ DESC <5000'
TCT-1	20	9	21	56	23
TCT-2	–	–	2	107	11
TCT-3	6	12	19	111	36
•	•	•	•	•	•
TCT-5	51	20	70	–	12
Total	779	330	604	2098	402

- Avg. Flt Time = 3.19 Hours With 1.6 Hours at Low Alt (< 5000 Ft)
- 30.8 Flights/Training Cycle
- Flight Time/Cycle = 30.8 x 3.19 = 98.2 Hrs

CLIMATIC DATA

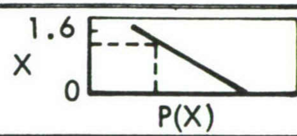
EDWARDS AFB AREA

EGLIN AFB AREA

	Days of Rain	Inches of Rain
Jan	9	3.86
Feb	9	4.0
•	•	•
•	•	•
Dec	9	4.16
Total	112	57.61

- Avg Amount of Rain on Rainy Day $\frac{57.12}{112} = 0.5$ In.
- Assume Rainfall Rate of 0.1 In./Hr.
- Length of Rain $\frac{0.5}{0.1} = 5.0$ Hrs.

PROBABILITY OF FLYING (X) HOURS IN RAIN



EXPECTED TIME IN RAIN PER FLIGHT

$$\frac{(1.4)(.01765) + (0.2)(.02885)}{2}$$

$$\frac{.03048}{2} = .0152 \text{ Hrs.}$$

AIRCRAFT UTILIZATION

- 25/Hrs/Crew/Month Required
- 100 Crews
- 72 Aircraft
- Aircraft Hrs. Per Mo. $25 \times \frac{100}{72} = 35$ Hrs

- Probability of Rain $\frac{112}{365} = .31$
- Probability of Flying $\frac{30.8/98.2/35}{365/12} = .36$
- Probability of Flt on Rainy Day $.31 \times .36 = .1116$

FLIGHT EXPOSURE PER TRAINING CYCLE

$.0152 \times 30.8 =$ $.46$ Hrs/Cycle

Figure 5-2

The probability of flying (x) hours in rain can be determined by the equation

$$P_{(x)} = \left(\frac{T_R - \left[x-(T_F-x)\right]}{T}\right)\left(p_r\right)$$

where T_R = average number of hours of rain

x = number of hours flown in the rain

T_F = hours in the training flight at altitude <5000 feet

T = hours in a day

p_r = probability of flight on a rainy day

Substituting the values shown in Figure 5-2 in the above equation, the probability of flying (x) hours in the rain can be computed as

$$P_{(1.4)} = \left(\frac{5.0-\left[1.4-(1.6-1.4)\right]}{24}\right)(.1116) = \left(\frac{3.8}{24}\right)(.1116) = .01765$$

$$P_{(0.2)} = \left(\frac{5.0-\left[0.2-(1.6-0.2)\right]}{24}\right)(.1116) = \left(\frac{6.2}{24}\right)(.1116) = .02885$$

The expected aircraft flight exposure to rain during the training cycle can be determined as

$$F_R = \left[\frac{(1.4)(.01765)+(0.2)(.02885)}{2}\right]\left[30.8\right]$$

$$= \left(\frac{.03048}{2}\right)(30.8)$$

$$= 0.46 \text{ hours per training cycle}$$

The average relative humidity encountered during the training flights is 75 percent - the yearly average for the Eglin AFB flight ranges. The average daily humidity cycle, which was discussed for ground environment, is not considered a significant factor for the flight environment since flights over the range could occur at any time during the daily period.

Humidity condensation is another factor that must be considered in the flight environment. This condensation can be expected to occur when descending from high-altitude flight into the humid air at low altitudes. The temperatures of the aircraft surfaces, which have been lowered by exposure to the colder air at the higher altitudes, are below the dewpoint temperatures of the low-altitude air. Thus, condensation on the surfaces will occur until their temperatures respond to the existing ambient temperatures and climb above the dewpoint. Exposure of the aircraft to this environment was not determined; however, the critical parts are subjected to the same sequence of events, and their exposure to condensation is established in the following section.

5.3 Aircraft Environment. The inherent aircraft environments which are considered because of their possible effects on critical parts are fuel, lubricant and hydraulic fluid. While the aircraft are on the ground at their bases, they are normally kept fully fueled in accordance with Air Force operating procedures; therefore, all "tankage" areas would be exposed to a fuel environment during this period.

In flight, the fuel management of the F-111 discussed in Reference 8 indicates that the fuel usage sequence is the wing tanks first, then the fore and aft fuselage tanks, and finally the reservoir tank. The initial decrease in aircraft gross weight, therefore, indicates the usage of wing fuel.

The methodology used to determine the flight time that fuel is in the wing tanks is shown in Figure 5-3. The aircraft gross weight versus flight time is plotted in Reference 4, and the flight time at which aircraft gross weight drops an amount equal to the wing fuel weight is the expected exposure to a fuel environment in the wing tanks. The times that fuel

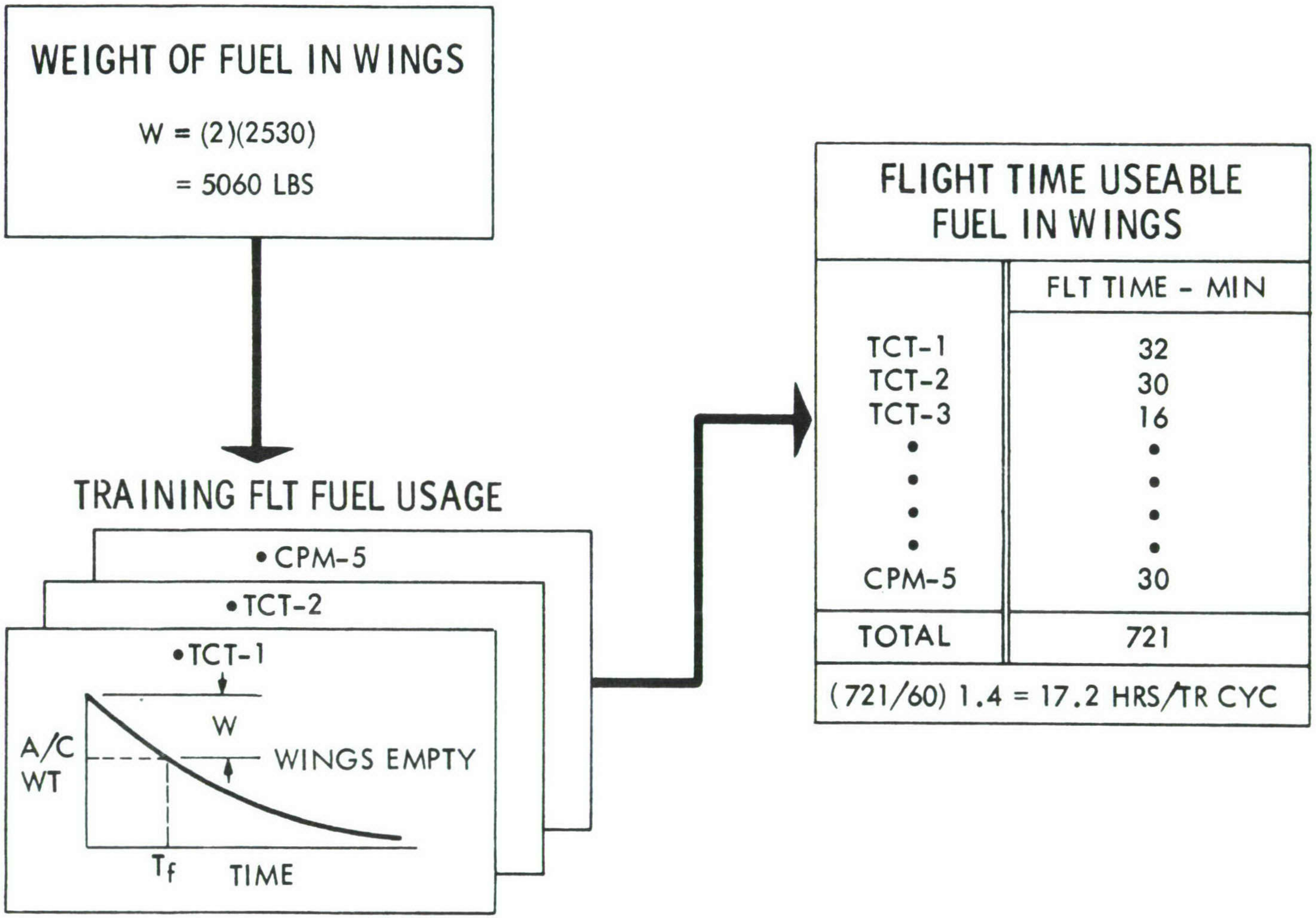


Figure 5-3 EXPOSURE TO FUEL DURING FLIGHT –
INNER SURFACES OF WING PIVOT FITTING PLATES & WEBS

is in the wing tanks for each of the training flights are listed in Table 5-1. The total flight time with fuel in the wing tanks is 721 minutes or 12.0 hours for the 22 flights, and 17.2 hours for the 30.8 flights during the training cycle.

It should be noted, however, that about ten pounds of unuseable fuel remains in each wing. Pressurization and vibration during flight will normally restrict this fuel to the bottom of the tank; but it is probable that the fuel will randomly immerse all parts in the wing tanks due to the motion of the aircraft.

As previously noted, the fuel in the reservoir tank is the last to be used. Since landing reserve requirements exceed the fuel capacity of this tank, it is expected that a fuel environment will exist in the reservoir tank during the entire flight.

TABLE 5 -1 Flight Time With Fuel in Wing Tanks

Training Mission	Flight Time With Wing Fuel-Minutes
TCT-1	32
TCT-2	30
TCT-3	16
TCT-4	30
TCT-5	68
TCT-6	30
TCT-7	30
TCT-8	30
TCT-9	30
TCT-10	30
TCT-11	30
TCT-12	36
TCT-13	23
TCT-14	Dry Wing
TCT-15	74
TCT-16	27
TCT-17	23
CPM-1	38
CPM-2	38
CPM-3	27
CPM-4	49
CPM-5	30
TOTAL	721

The wing pivot on the F-111 is lubricated every 50 flight hours or 5 wing sweeps on the ground to provide minimum wear on the pivot fittings. The lubricant used for the pivot consists of a poly-alkaline base with a molybdenum disulfide filler and is referred to by its specification number FMS-1071 (MoS_2-1195 in Reference 9).

The hydraulic system on the F-111 is another possible source of an environment which is inherent in the aircraft. The presence of hydraulic fluid from the system would most likely be found in the vicinity of the actuators as a result of the accumulated leakage from both normal actuator operations and maintenance actions. The hydraulic fluid is Mil-H-5606 which has a petroleum base with additives to provide coloring as well as anti-foaming and corrosion-inhibiting characteristics.

5.4 Environment Induced from Ground Operations. The major environments induced on the aircraft from ground operations result from washing and deicing. The aircraft must be washed at least every 30 days according to Reference 10, and more often if required. The aircraft is washed with a mixture of water and Mil-C-25769 cleaning compound which contains silicates, phosphates and a wetting agent. This cleaning compound is 90 percent biodegradable.

Flight operations during snow and/or ice conditions can impose two requirements on ground operations according to Reference 10. First, aircraft deicing (or snow removal) before flight requires the use of Mil-A-8243 solvent which is an ethylene glycol-propolene glycol mixture. This solvent is preferred over Federal Spec P-D-680, Type II solvent.

Secondly, the aircraft must be cleaned after the flight to remove the corrosive material used for deicing the runways. The aircraft can be cleaned by spraying with a mixture of water and Mil-C-27251 cleaning compound, brushing, and rinsing with MIL-A-8243 solvent.

6.0 ENVIRONMENTAL EXPOSURE OF CRITICAL PARTS

The chemical environments on the F-111s based at Homestead AFB, Florida were developed in the preceding section. The location of the critical parts on the aircraft must be established. These data are then analyzed to determine the exposure of the critical parts to the different environments.

6.1 Location of Critical Parts. The 15 critical parts are depicted in Figure 6-1 along with their location in the aircraft. The locations of these parts in combination with the locations of access panels and drain holes as illustrated in Reference 11 provide a good insight to the potential paths which allow exposure of the critical parts to the external environment on the aircraft.

An evaluation of the above information indicated that while the aircraft was on the ground, rain would penetrate around the access covers and come in contact with the critical parts. Also, humid air can reach the parts through the access covers and drain holes although there would be very little internal circulation of this air. In flight, rain and humid air would reach the critical parts through the access covers.

Humidity condensation on the external aircraft surfaces was adjudged not to be a significant part exposure factor either on the ground or in flight. On the ground, this external condensation does not create sufficient quantities of moisutre on the upper surfaces of the aircraft to flow through access covers and reach the parts. In flight, the exposure due to condensation on the critical parts is a more significant measure than condensation on external aircraft surfaces.

6.2 Ground Exposure. The ground exposure of the critical parts considered in this study is the summation of exposures to rain, different ranges of relative humidity, condensation of humid air, lubricant and fuel.

The first three environments listed above are mutually exclusive. The exposure times to the different humidity ranges are exclusive of condensation time. Humidity condensation, however, can occur when the relative humidity is in either of

LOCATIONS OF 15 CRITICAL FORGED-STEEL PARTS ON F-111

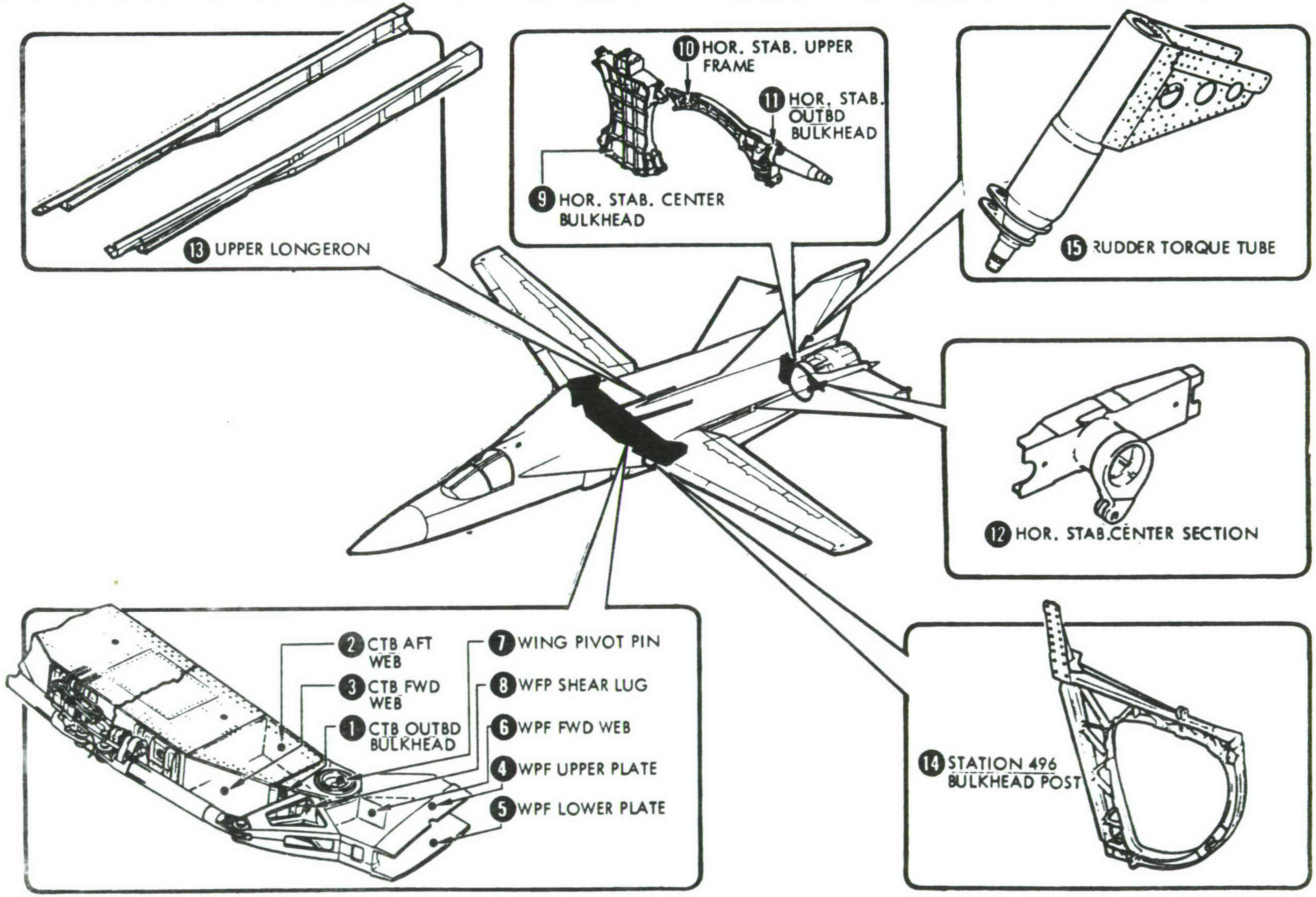


Figure 6-1

the higher ranges; this is taken into account by categorizing this part of the exposure as humidity condensation which results in a corresponding reduction in the exposure estimate applied to the latter two humidities. The sum of the exposure times to these environments is equal to the total time the aircraft is on the ground during the training cycle. The above relationships can be expressed by the equation

$$E_G = E_R + E_{H1} + E_{H2} + (E_{H3} + E_{H4} + E_C)$$

where

E_G = total ground exposure time per training cycle

E_R = exposure time to rain

E_{H1} = exposure time to relative humidity in the 0-25 percent range

E_{H2} = exposure time to relative humidity in the 25-50 percent range

E_{H3} = exposure time to humidity in the 50-75 percent range

E_{H4} = exposure time to humidity in the 75-100 percent range

E_C = exposure time to humidity condensation

The total ground exposure time per training cycle is 1947.8 hours as determined in Section 3.0. The rain environment on the aircraft at Homestead is 134.3 hours as developed in Subsection 5.1, and the exposure of the critical parts to rain is also 134.3 hours based on the evaluation of critical part location. Substituting this exposure in the above equation, it can be seen that there are 1813.5 hours remaining in the training cycle for exposure to the other environments.

The analysis of the average daily relative humidity cycle in Subsection 5.1 revealed that the relative humidity was always above the 0-25 percent ranges; therefore, there was no part exposure to such environments. The relative humidity was in the 50-75 percent ranges 45.8 percent of the time, or 830.6 hours of exposure for the critical parts.

The 75-100 percent range for relative humidity occurred 54.2 percent of the time, or 982.9 hours of part exposure. During the period of higher relative humidity, however, condensation of the humid air can occur.

Condensation of humid air can normally be expected to occur in the pre-dawn hours when the relative humidity is the highest and the cumulative heat loss from a radiating body is the greatest. An illustration of the resulting temperature relationships is shown in Figure 6-2.

The temperature plots shown in Figure 6-2 represent the ambient temperature, the dewpoint temperature (a function of the relative humidity), and the aircraft part temperature during the pre-dawn period when condensation is most likely to occur. After the sun sets, the relative humidity starts to increase which corresponds to the dewpoint temperature dropping more slowly than the ambient temperature. At the same time, the aircraft part temperature is dropping faster than the ambient due to heat loss through radiation. These temperature trends can continue until the part temperature drops below the dewpoint temperature. During the period that this condition occurs, moisture condenses on the part from the air. The condensation period ends when these temperature trends are reversed, usually the sunrise which tends to lower the relative humidity and reverse the radiation process on the parts.

Although heat loss through radiation is normally associated with external parts or surfaces, internal aircraft parts are also subject to heat loss during this period through a combination of radiation, convection and conduction. For the aircraft based at Homestead, the combination of relative humidity and estimated part temperatures during the average daily cycle resulted in exposure of the critical parts to 170.5 hours of condensation during the training cycle. After establishing this condensation exposure time, it can be determined that the critical parts are exposed to the 75-100 percent humidity for 812.4 hours.

The maximum condensation on the ground occurs during the pre-dawn period when there is a minimal probability of takeoffs; also there is a very low probability of takeoff during rain. Any moisutre remaining on the parts after such periods will likely be removed by avionics and engine warmup, taxi and takeoff. The rain and condensation which occur on the ground, therefore, are considered to present a negligible residual moisture environment on the critical parts during flight.

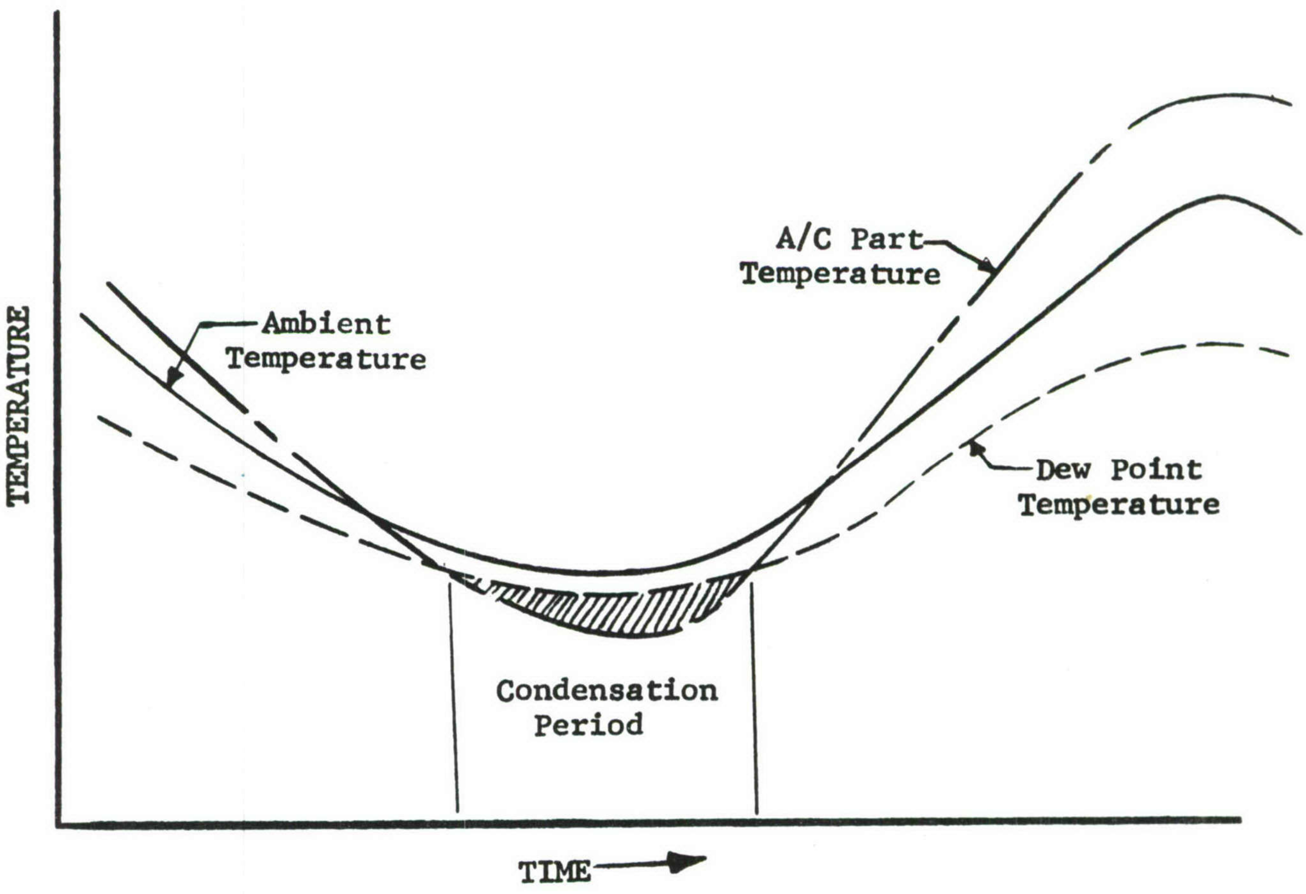


Figure 6-2 Illustration of Condensation Period in Daily Temperature Cycle

The ground exposure of certain critical parts to the fuel environment occurs continuously when the aircraft is on the ground during the training cycle. The parts exposed to fuel during this period are the internal surfaces of wing-pivot-fitting plates and webs, and the webs and bulkheads on the carry through box.

The wing pivot pin is at the pivot point of the wing which is kept lubricated at all times to minimize wear. The pivot pin, therefore, is exposed to the FMS-1071 lubricant for the 1947.8 hours on the ground during the training cycle.

The exposure of critical parts to the smoke and sulphur dioxide concentrations at Upper Heyford should be much less severe than the exposure of external surfaces. This occurs because the limited internal space around the critical parts allows only very small deposits during each humidity cycle. The external surfaces however, will be cleansed periodically by rain and/or the required washing while internal deposits of the contaminants can build up during this period. The internal surfaces should be closely watched for evidence of corrosion.

The salt concentrations in the air at Homestead AFB will have a similar effect. Although the level of these concentrations is not known, it is certain that they are sufficient to create corrosion problems.

Another environment which could effect some of the critical parts is hydraulic fluid which accumulates from the leakage of the actuators in the hydraulic system. The critical parts which are located in the proximity of actuators and could thus have a thin coating of hydraulic fluid on their surfaces are those parts on the carry through box and wing pivot fitting, the horizontal-stabilizer outboard bulkhead and center section, and the rudder torque tube assembly.

6.2 Flight Exposure. The flight exposure of the critical parts is to a large degree a direct function of the environments on the aircraft during flight. The environments of primary interest in this study which are encountered in flight are rain, relative humidity, condensation of humid air, dry air, lubricant and fuel. The first three environments listed above are mutually exclusive. The exposure time to humidity is exclusive of condensation time. Humidity condensation,

however, can occur during the humidity exposure encountered in flight; this is accounted for by determining the exposure time to humidity condensation, and making a corresponding reduction in the exposure time to humidity. The sum of the exposure times to these environments is equal to the flight time when the aircraft is below 5000 feet during the training cycle. The above relationships can be expressed by the equation

$$F_T = F_R + (F_H + F_C)$$

where

F_T = total flight exposure time at low altitude

F_R = flight exposure time to rain

F_H = flight exposure time to relative humidity

F_C = flight exposure time to humidity condensation.

The total flight exposure at low altitudes during the training cycle is 49.3 hours as developed in Section 3.0. The rain exposure expected on the Eglin flight ranges for the critical parts is 0.46 hours - the same as the rain exposure determined for aircraft in Section 5.2. The critical part and overall aircraft exposure to rain are the same as explained previously in this section. Substituting the rain exposure into the above equation, it can be seen that humidity exposures during the training cycle are 48.8 hours.

The part exposure to condensation of humid air during flight was determined as shown in Figure 6-3. The relative humidity of 75 percent on the Eglin AFB flight ranges results in a dewpoint temperature which is 9°F below the ambient temperature. The flight profiles of Reference 6 and the part temperature analyses of Reference 12 were employed to estimate the temperature of the critical parts due to altitude changes during the training flights. These data were compared to determine the time that the part temperatures were below the dewpoint temperatures, and thus the time that moisture was condensing on the parts. The condensation times are listed in Table 6-1, for each training flight. The total part

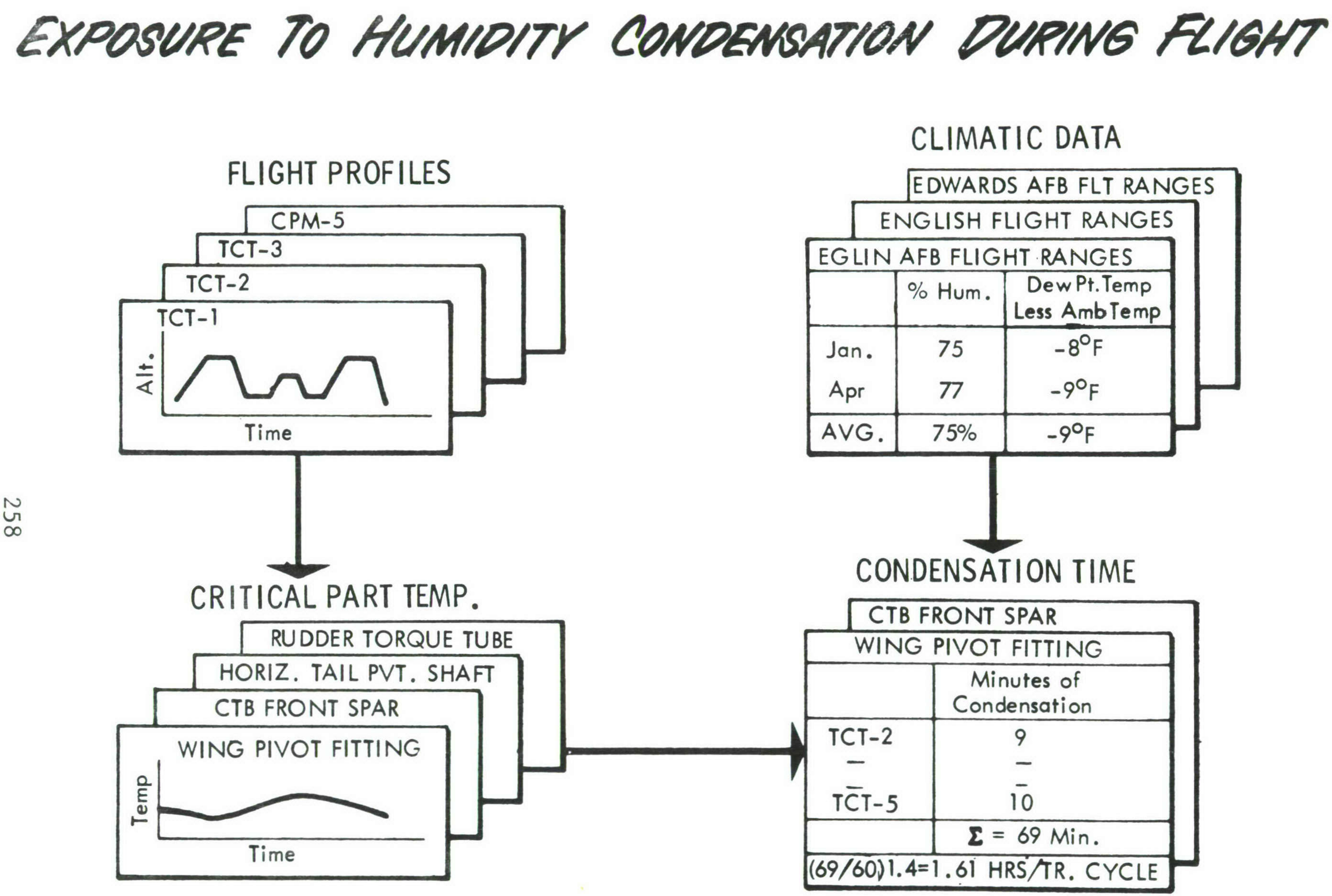

EXPOSURE TO HUMIDITY CONDENSATION DURING FLIGHT
FLIGHT PROFILES
CPM-5
TCT-3
TCT-2
TCT-1
Alt.
Time
CLIMATIC DATA
EDWARDS AFB FLT RANGES
ENGLISH FLIGHT RANGES
EGLIN AFB FLIGHT RANGES
% Hum.
Dew Pt. Temp Less Amb Temp
Jan. 75 -8°F
Apr 77 -9°F
AVG. 75% -9°F
CRITICAL PART TEMP.
RUDDER TORQUE TUBE
HORIZ. TAIL PVT. SHAFT
CTB FRONT SPAR
WING PIVOT FITTING
Temp
Time
CONDENSATION TIME
CTB FRONT SPAR
WING PIVOT FITTING
Minutes of Condensation
TCT-2 9
TCT-5 10
Σ = 69 Min.
(69/60)1.4=1.61 HRS/TR. CYCLE

Figure 6-3

TABLE 6-1 Condensation Times on Critical Parts During Flight on Eglin AFB Ranges

Training Mission	Condensation Time - Minutes
TCT-1	0
TCT-2	9
TCT-3	0
TCT-4	0
TCT-5	10
TCT-6	0
TCT-7	0
TCT-8	0
TCT-9	0
TCT-10	0
TCT-11	0
TCT-12	10
TCT-13	0
TCT-14	17
TCT-15	6
TCT-16	2
TCT-17	0
CPM-1	0
CPM-2	0
CPM-3	2
CPM-4	13
CPM-5	0
TOTAL	69

exposure to condensation in the above table is 69 minutes or 1.15 hours for the 22 flights, and 1.61 hours for the 30.8 flights per training cycle.

Subtracting this condensation time from the total time of exposure to humidity leaves 47.2 hours. The flight exposure time to the relative humidity of 75 percent, then, is 47.2 hours.

The wing pivot pin is exposed to the FMS-1071 lubricant all the time, or 98.2 hours per training cycle during flight. The exposure to fuel during flight is 17.2 hours for the critical parts on the wing pivot fitting (WPF), and 98.2 hours for the parts on the carry through box (CTB) since it is part of the reservoir tank. These times were established in Section 5.2. The critical parts on the WPF are the upper and lower plates, and the forward web; the parts on the CTB are the forward web, the aft web and the outboard bulkhead.

7.0 REFERENCES

1. Memo to W. D. Buntin (from E. G. Brooker/R. J. Green), Subject: Chemical Exposure of D6ac Steel Parts During Phase III Flight Training, dated 26 January 1971.

2. F-111A, E, D, & F Wing Squadron Allocation and Support Plan, General Dynamics/Fort Worth Division, Industrial Engineering and Scheduling, Schedule No. 12-0-76J, dated 7 May 1970.

3. Note to James S. Hall (from Lt. Colonel Ken Blank of 20th TFW, Upper Heyford, U.K.), Subject: F-111 Operations in USAFE, dated 3 December 1970.

4. Lowrey, B. T.; Lillie, R. B.; and Cetola, R. T.; F-111A/E/D Mission Analysis to Determine Maneuver Load Factor Exceedance Spectra, General Dynamics/Fort Worth Division, FZM-12-10783, dated 27 June 1969.

5. Climatic Data Summaries, Fifth Weather Wing, Langley AFB, Virginia.

6. Letter to B. A. Kress (from D. L. R. Bailey of Warren Spring Laboratory in Hertfordshire, England), Subject: Air Pollution Measurements in the Upper Heyford Area, dated 10 November 1970.

7. Climatic Atlas of the United States, U. S. Department of Commerce, Environmental Services Administration, Environmental Data Service, dated June 1968.

8. F-111E Flight Manual, U. S. Air Force, T.O. 1F-111E-1, dated 6 June 1969.

9. F-111 Wing Pivot Joint - Lubrication System Development Program, General Dynamics/Fort Worth Division, FZM-12-958, dated 26 February 1965.

10. Cleaning of Aerospace Equipment, U. S. Air Force, T.O. 1-1-1, dated 5 May 1969.

11. Corrosion Control - F-111E, U. S. Air Force, T.O. 1F-111E-23, dated 26 June 1970.

12. F-111A D6ac Steel Critical Part Temperature Study, General Dynamics/Fort Worth Division, FZM-12-10946 (Appendix B) dated 30 January 1970.

APPENDIX A

IMPLICATIONS OF CURRENT F-111 OPERATIONS ON ENVIRONMENTAL EXPOSURE DATA

The analysis documented in this report was completed in January 1971, and the results on environmental exposure of F-111 airframe components were factored into analyses being conducted to establish appropriate inspection intervals for critical D6ac steel parts. These studies were initiated at the request of the Scientific Advisory Board (SAB) which was commissioned to investigate the structural integrity of F-111 aircraft.

During the period that the environmental exposure analysis was in progress, the F-111F wing was planned to be based at Homestead AFB, Florida. Subsequently, basing for the F-111Fs was changed to Mountain Home AFB, Idaho. The climate at this latter base and the associated flight training ranges provides a much drier environment in terms of both humidity and rain. Therefore, compared to that predicted in the analysis, the critical parts on these aircraft are exposed to considerably less moisture both in flight and on the ground.

Another change which has considerable effect on the environmental exposure results is that the F-111 operations associated with the continuation flight training programs have been reduced compared to what was originally estimated. Later information indicates reductions in key parameters of the training programs, i.e., crew-to-aircraft ratio, aircraft utilization per month, etc. As a result, the flight time as a function of calendar time presently is less which reduces the environmental exposure rate for the critical parts on the aircraft under flight conditions.

A review of the above flight training changes reveals that the critical D6ac steel parts on the F-111s currently are subjected to lower exposure rates to environments during flight than was expected at the time the environmental analysis was conducted. The data contained in this report and used in the studies for the SAB Committee are therefore considered conservative, and additional analyses to determine current environmental exposure data are not required.

SUPPLEMENT (B)

F-111A D6Ac STEEL CRITICAL PART

TEMPERATURE STUDY

F-111A

D6AC STEEL CRITICAL PART TEMPERATURE STUDY

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1. INTRODUCTION

This report contains the results of a survey made to determine minimum temperatures of D6AC steel parts as installed and operated on the F-111 airplane. Mission profiles, temperature-altitude conditions, part locations, and temperature histories are given.

2. SCOPE

2.1 Parts Considered

Airplane parts, and specific locations on parts for this study were obtained from Reference (1). Critical locations selected were on the wing pivot fitting plate and the wing pivot fitting lug (12W473), on the F.S. 496 Bulkhead (12B2910), on the upper longeron (12B1891), and on the horizontal tail pivot shaft (12B10521).

2.2 Missions Considered

Missions used in this study were selected from TAC Syllabus, Course 111508C (Reference 2), and from 111 performance data. The selection criteria were minimum exposure temperature and maximum exposure time at low temperature. From Reference (2), missions TR(A)-2 and TR(A)-5 were selected, with the modification of deleting the $M = 1.5$ operation from TR(A)-2. Airplane performance and manuever capabilities were used to determine a maximum endurance loiter condition and the tracjectory of a subsequent 20^{o} dive with 4-g pullout. A max range ferry mission was also obtained. These missions are shown on Figures 7 and 8.

2.3 Atmospheres

The variations of temperature with altitude (i.e, "Day") for this study were obtained from MIL-STD-210A, Reference (3), and from AFCRC-TR-59-267, Reference (4). These are referred to in this report as MIL-STD-210A Polar Day, MIL-STD-210A Tropical Day, and ARCD1959 Standard Day. Plots of adiabatic wall temperature vs Mach Number and altitude for these atmospheres are given on Figures 9, 10, and 11, respectively.

3. METHOD OF ANALYSIS

Transient temperatures predictions for this study were made using General Dynamics IBM 360 Computer Program RT2. This program uses standard lumped-parameter finite-difference techniques to determine transient temperature distributions. Radiation heat losses to a -40°F sky temperature were considered, with a surface emittance of 0.8. Aerodynamic heat transfer coefficients were computed by the program for external surfaces. Surfaces exposed to nacelle secondary air or engine inlet air were assumed to have heat transfer to ram air temperature, with a heat transfer coefficient of 20 Btu/hr-ft^2-°F.

4. COMPARISON WITH FLIGHT TEST DATA

Flight test data from F-111A No. 6 (Tail No. 63-9771), were obtained by NASA in August of 1967. Figure 62 shows flight path data from Flight 19, flown August 16, 1969. This flight was selected as having the most nearly continuous flight path data. Figure 63 shows 4 temperatures recorded on the vertical tail skin, 770 frame near the horizontal tail pivot shaft, wing splice temperature, and wing pivot fitting lug temperature. Figure 64 shows a comparison of the recorded wing splice temperature with RT2 predictions for the flight profile of Figure 62. On Figure 64, TEMP 3 is a value obtained from a one dimensional analysis of the steel plate, inboard of the splice. This reproduction of flight test temperature indicates that the analysis can be used to determine wing temperatures for other flight profiles.

5. PREDICTED TEMPERATURES

Figures 12 through 21 show temperature of the wing pivot plate for the four missions considered, for Standard, Polar, and Tropical atmospheres. Figures 12 through 14 are for Mission TR(A)-2, for the three days, Figures 15 through 17 are for Mission TR(A)-5, for the three days, Figures 18 through 20 are for the 20^{o} Dive from Loiter for the three days, and Figure 21 is the Ferry Mission for a Polar Day. Figures 22 through 31 are similarly arranged for the complete wing lug and plate results. Figures 32 through 41 are for the 496 bulkhead, Figures 42 through 51 are for the upper longeron, and Figures 52 through 61 are for the Horizontal Tail Pivot Shaft.

6. REFERENCES

1. FZM-12-10936, Report to USAF Scientific Advisory Board AD HOC Committee on F-111, 17 January 1970.

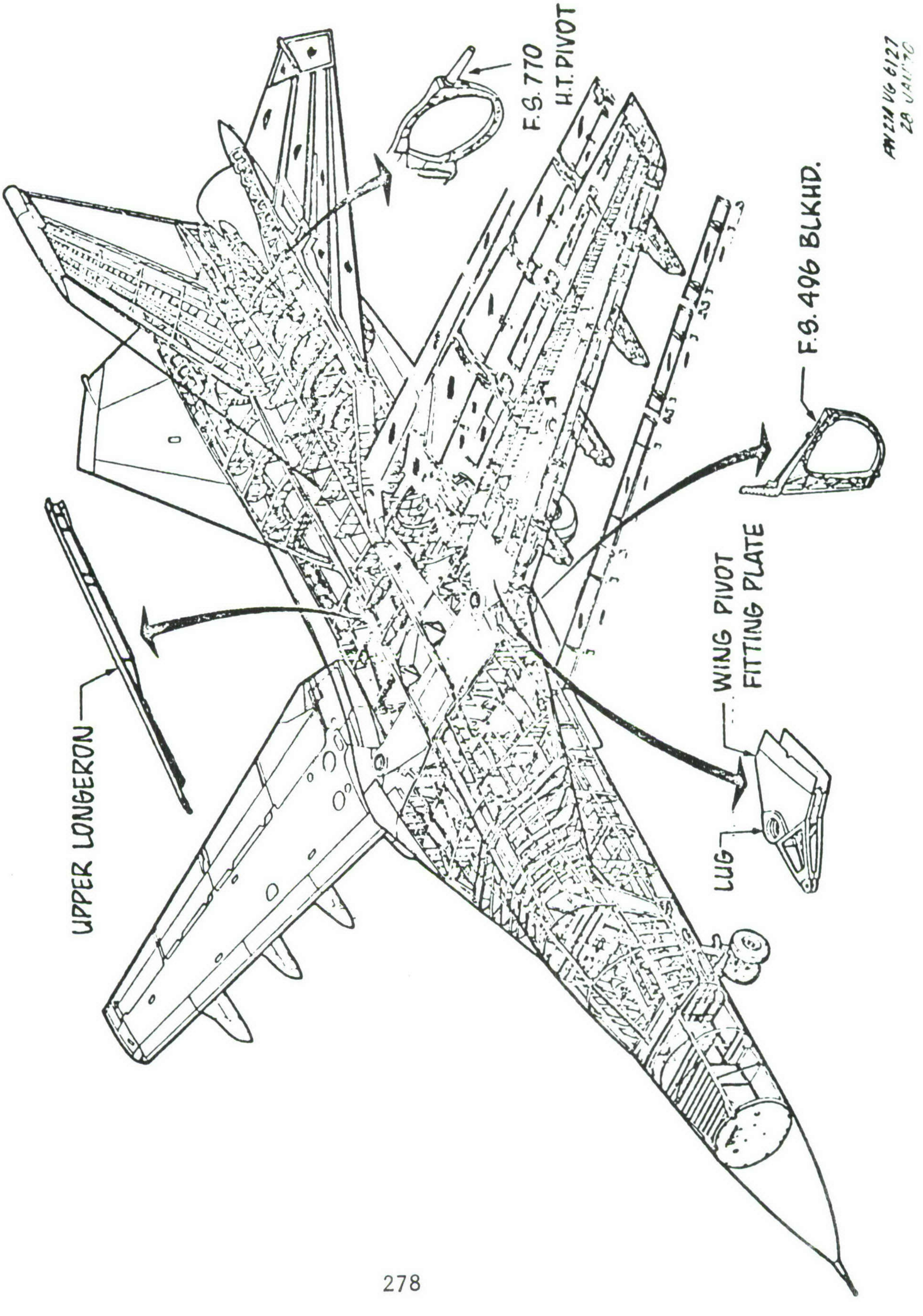
2. TAC Syllabus, Course 111508C, USAF Operational Training Course, F-111A.

3. MIL-STD-210A, Climatic Extremes for Military Equipment, 29 August 1957.

4. AFCRC-TR-59-267, The ARDC Model Atmospheres, 1959.

Figure 1

STRUCTURE COMPONENTS SELECTED FOR ANALYSIS



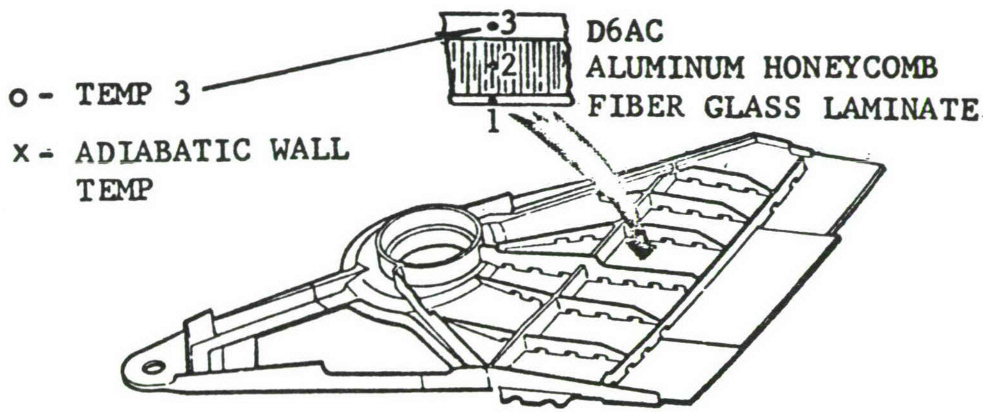


Figure 2 LOCATION OF CRITICAL TEMPERATURE NODES - WING PIVOT PLATE (PART NO. 12W473)

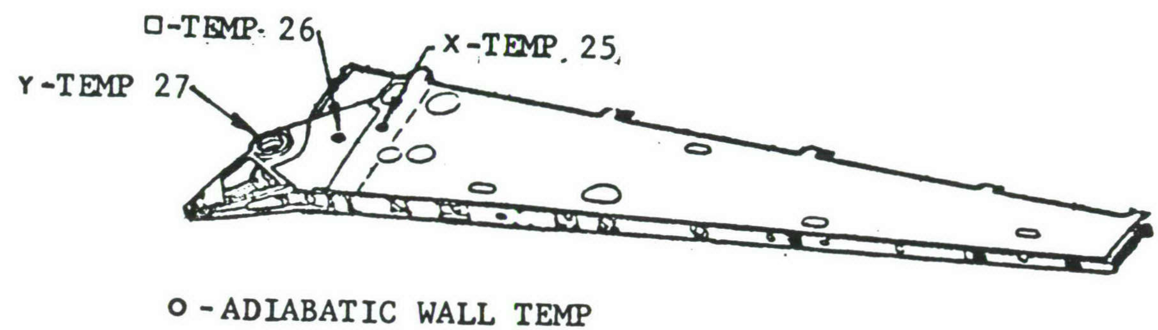


Figure 3 LOCATION OF CRITICAL TEMPERATURE NODES - COMPLETE WING APPROXIMATE ANALYSIS

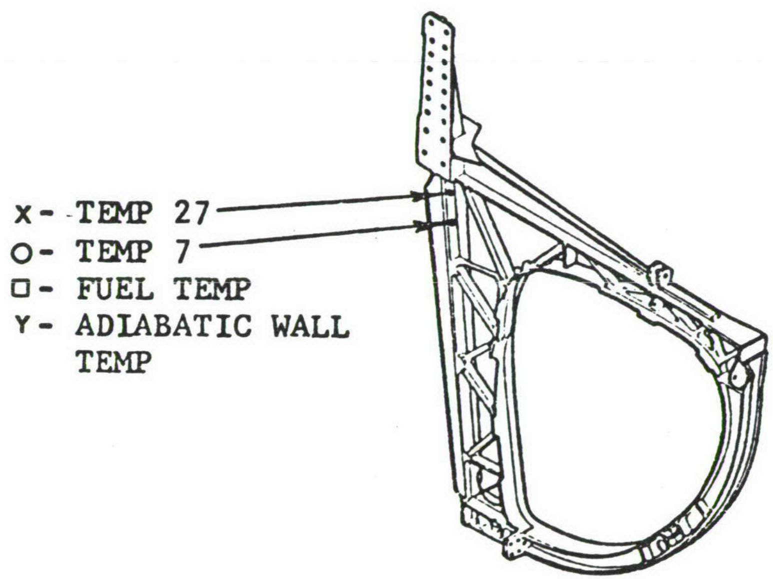


Figure 4 LOCATION OF CRITICAL TEMPERATURE NODES - F.S. 496 BULKHEAD (PART NO. 12W2910)

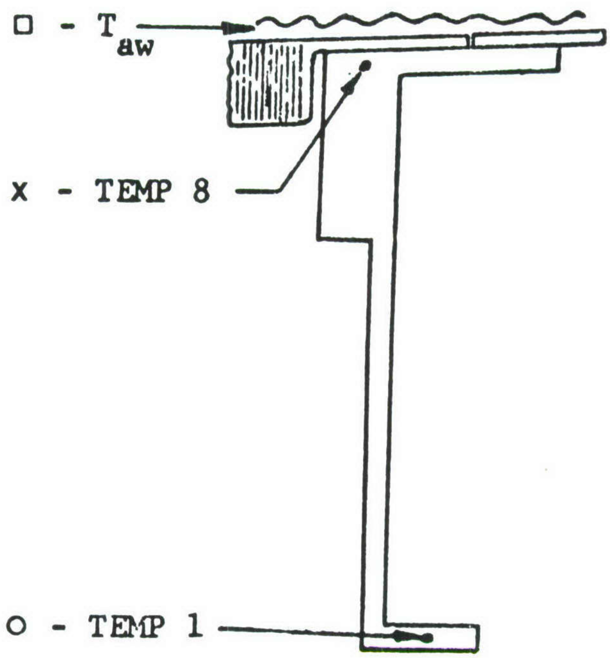


Figure 5 LOCATION OF CRITICAL TEMPERATURE NODES - UPPER LONGERON (PART NO. 12W1891)

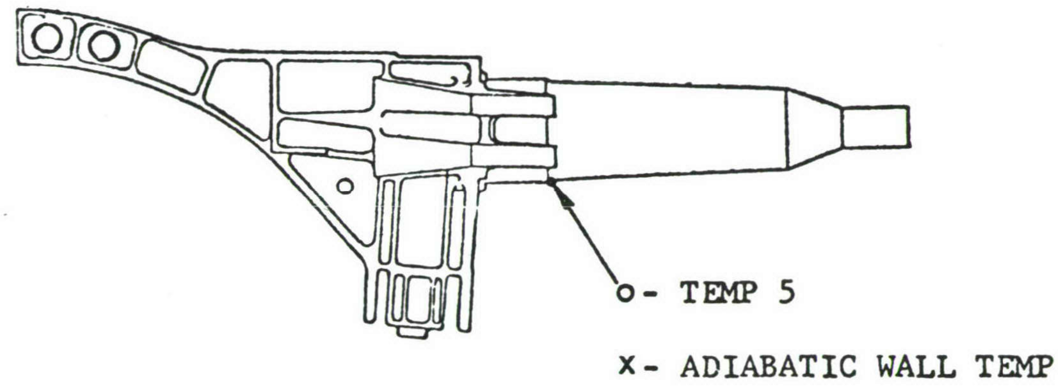
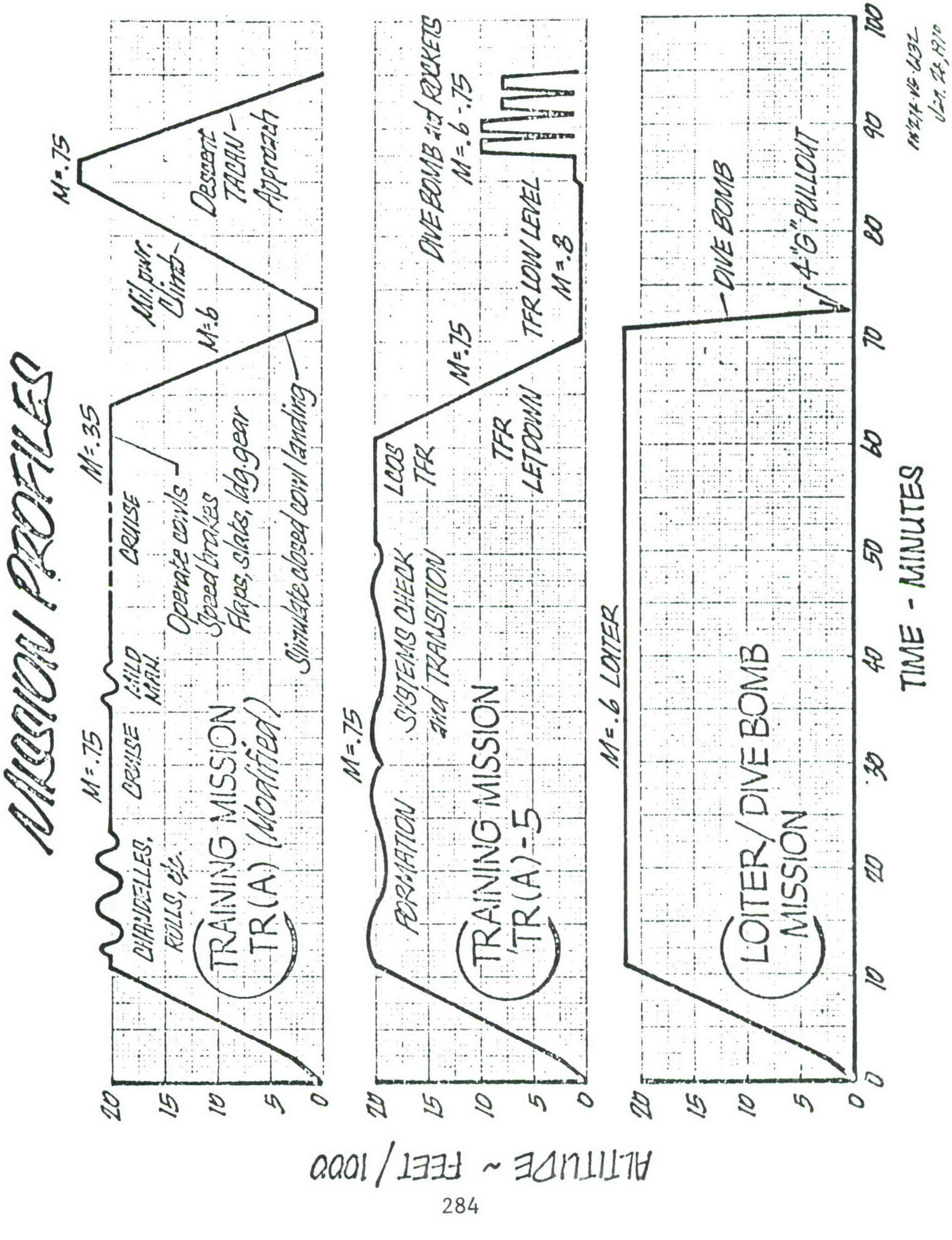


Figure 6 LOCATION OF CRITICAL TEMPERATURE NODES - HORIZONTAL TAIL PIVOT SHAFT, F.S. 770 BULKHEAD (PART NO. 12W10521)

Figure 7



MISSION PROFILES
TRAINING MISSION TR(A) (Modified)
CHANDELLES, ROLLS, etc.
M=.75
CRUISE
MILD MAN.
CRUISE
M=.35
Operate cowls
Speed brakes
Flaps, slats, ldg. gear
Simulate closed cowl landing
M=.6
Mil. pwr. Climb
M=.75
Descent
TACAN
Approach
TRAINING MISSION TR(A)-5
FORMATION
M=.75
SYSTEMS CHECK and TRANSITION
LCOS
TFR
TFR
LETDOWN
M=.75
TFR LOW LEVEL
M=.8
DIVE BOMB and ROCKETS
M=.6-.75
LOITER/DIVE BOMB MISSION
M=.6 LOITER
DIVE BOMB
4"G" PULLOUT
TIME - MINUTES
ALTITUDE ~ FEET / 1000

Figure 8

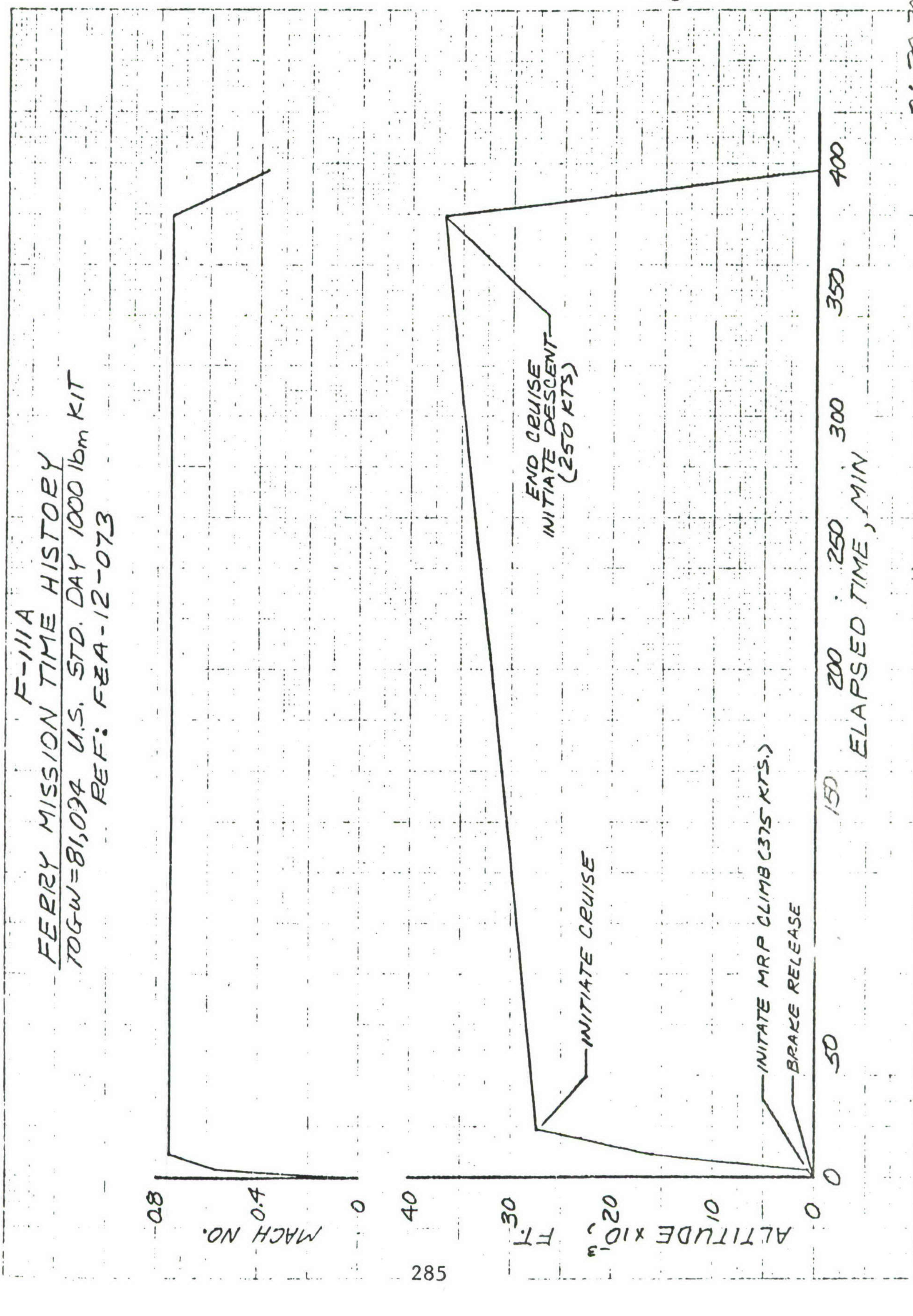

F-111A
FERRY MISSION TIME HISTORY
TOGW=81,094 U.S. STD. DAY 1000 lbm KIT
REF: FEA-12-073
END CRUISE
INITIATE DESCENT
(250 KTS)
INITIATE CRUISE
INITATE MRP CLIMB (375 KTS.)
BRAKE RELEASE
MACH NO.
0.8
0.4
0
ALTITUDE x10^-3, FT.
40
30
20
10
0
ELAPSED TIME, MIN
0
50
150
200
250
300
350
400
01-23-70

Figure 9

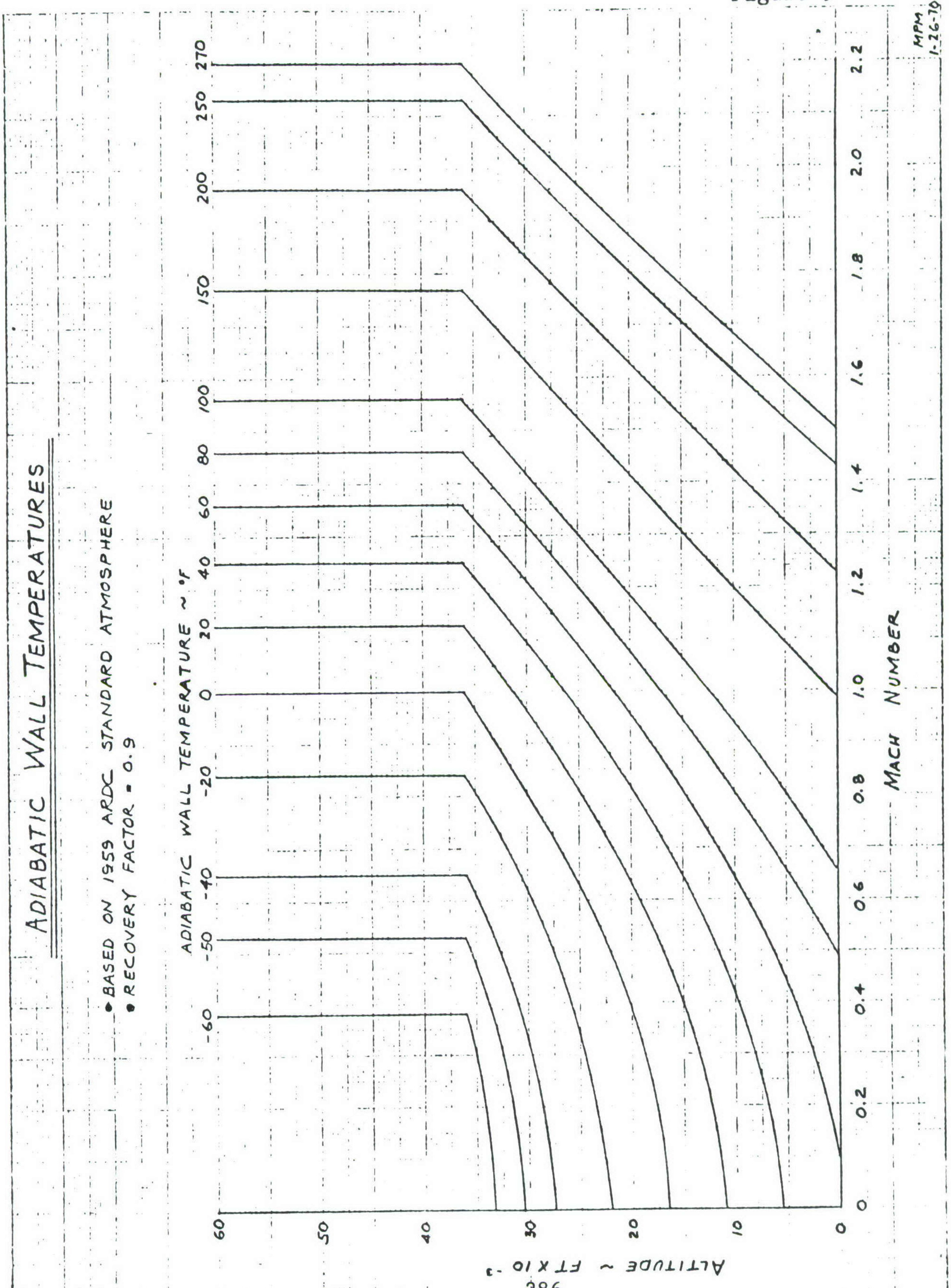

ADIABATIC WALL TEMPERATURES
• BASED ON 1959 ARDC STANDARD ATMOSPHERE
• RECOVERY FACTOR = 0.9
ADIABATIC WALL TEMPERATURE ~ °F
-60
-50
-40
-20
0
20
40
60
80
100
150
200
250
270
MACH NUMBER
0
0.2
0.4
0.6
0.8
1.0
1.2
1.4
1.6
1.8
2.0
2.2
ALTITUDE ~ FT X 10⁻³
0
10
20
30
40
50
60
MPM
1-26-70

Figure 10

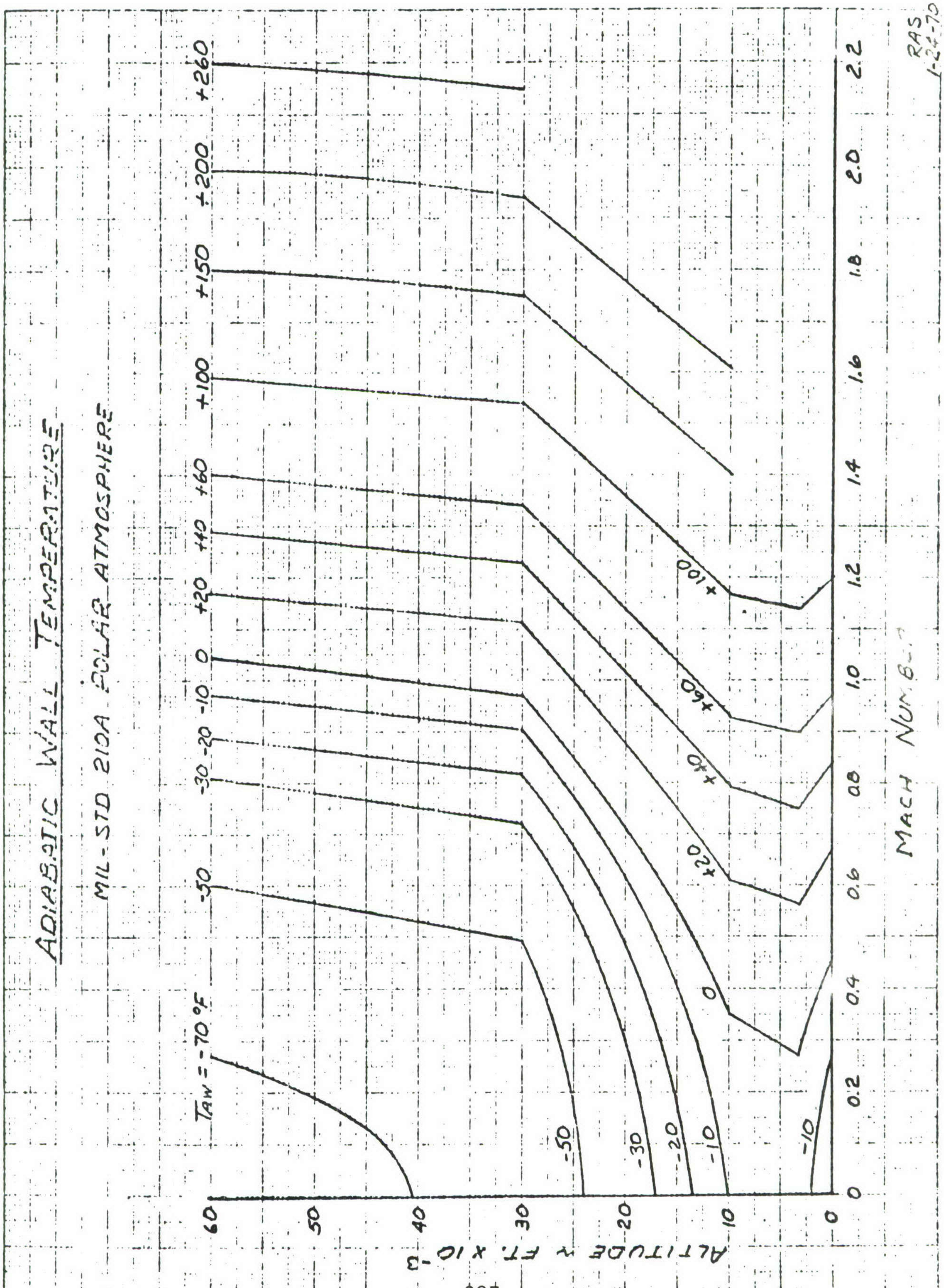

ADIABATIC WALL TEMPERATURE
MIL-STD 210A POLAR ATMOSPHERE
TAW = -70°F
-50
-30
-20
-10
0
+20
+40
+60
+100
+150
+200
+260
MACH NUM
0
0.2
0.4
0.6
0.8
1.0
1.2
1.4
1.6
1.8
2.0
2.2
ALTITUDE ~ FT. x 10^-3
60
50
40
30
20
10
0
RAS
1-24-70

Figure 11

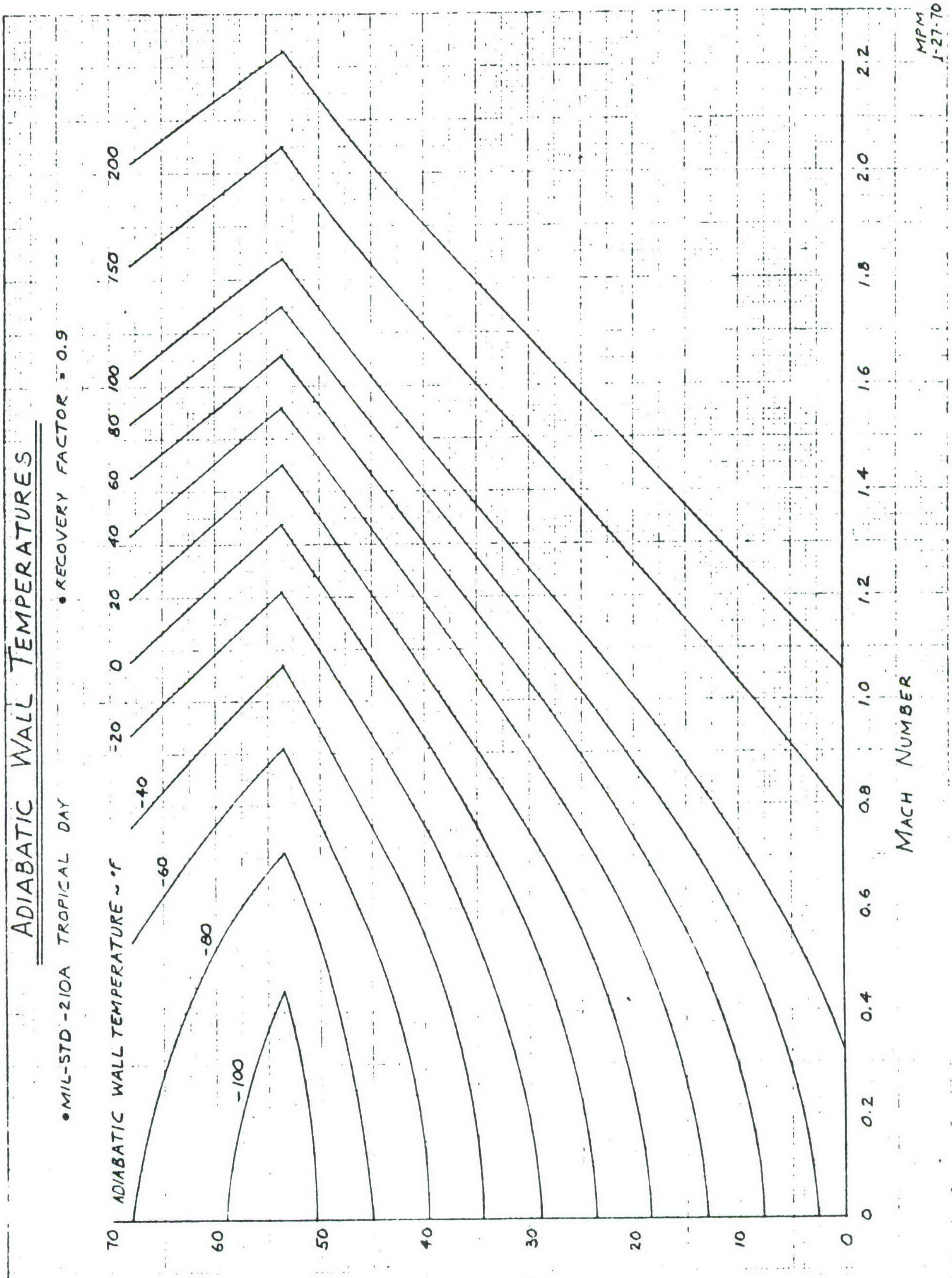

ADIABATIC WALL TEMPERATURES
•MIL-STD-210A TROPICAL DAY
•RECOVERY FACTOR = 0.9
ADIABATIC WALL TEMPERATURE ~ °F
200
150
100
80
60
40
20
0
-20
-40
-60
-80
-100
MACH NUMBER
0
0.2
0.4
0.6
0.8
1.0
1.2
1.4
1.6
1.8
2.0
2.2
ALTITUDE ~ FT X 10^-3
0
10
20
30
40
50
60
70
MPM
1-27-70

Figure 12

WING PIVOT FITTING 12W473
INBOARD OF SPLICE

F-111 TRAINING MISSION TR(A)-2
(M = 1.5 DELETED)

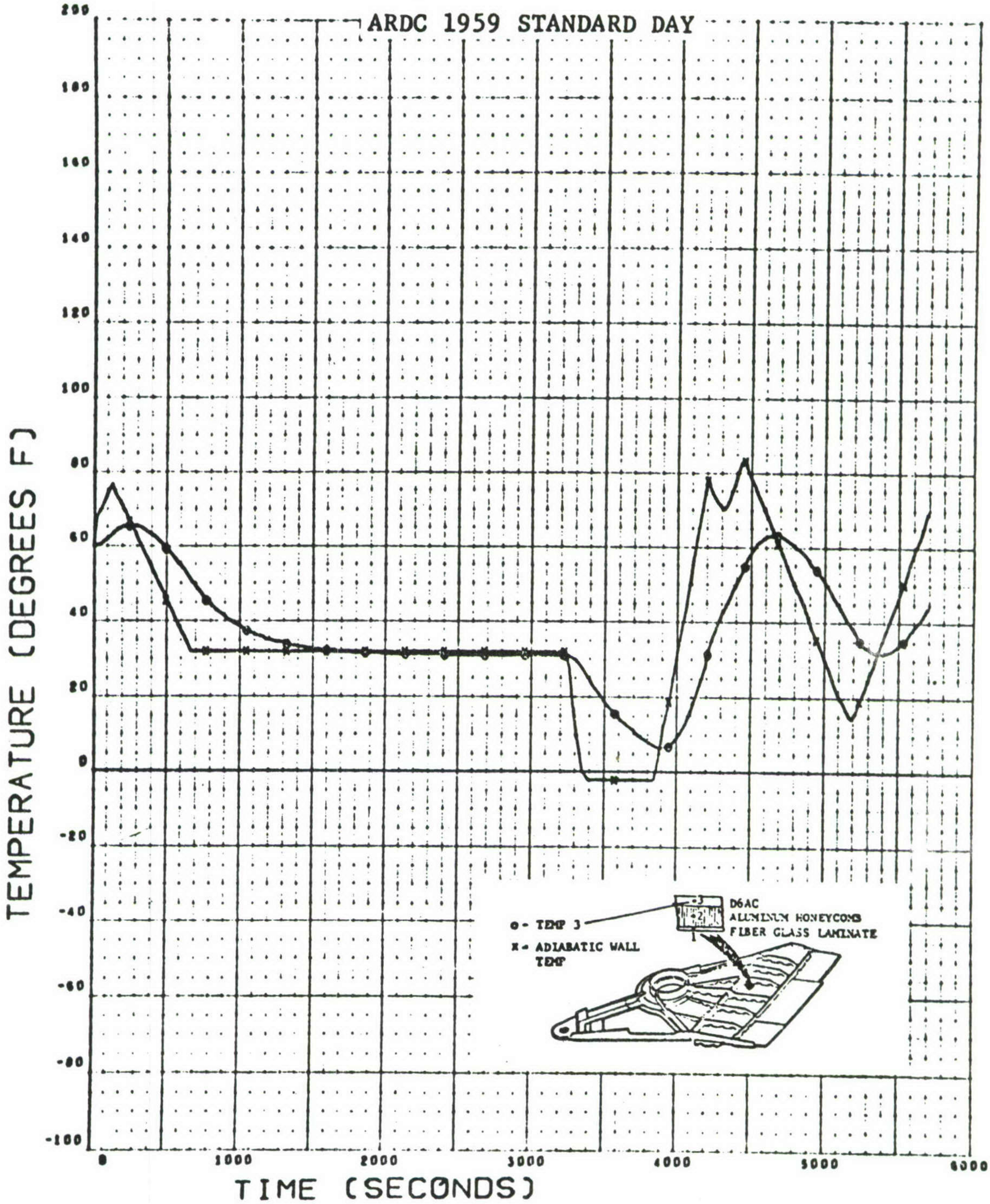


Figure 13

WING PIVOT FITTING 12W473
INBOARD OF SPLICE

F-111 TRAINING MISSION TR(A)-2
(M = 1.5 DELETED)

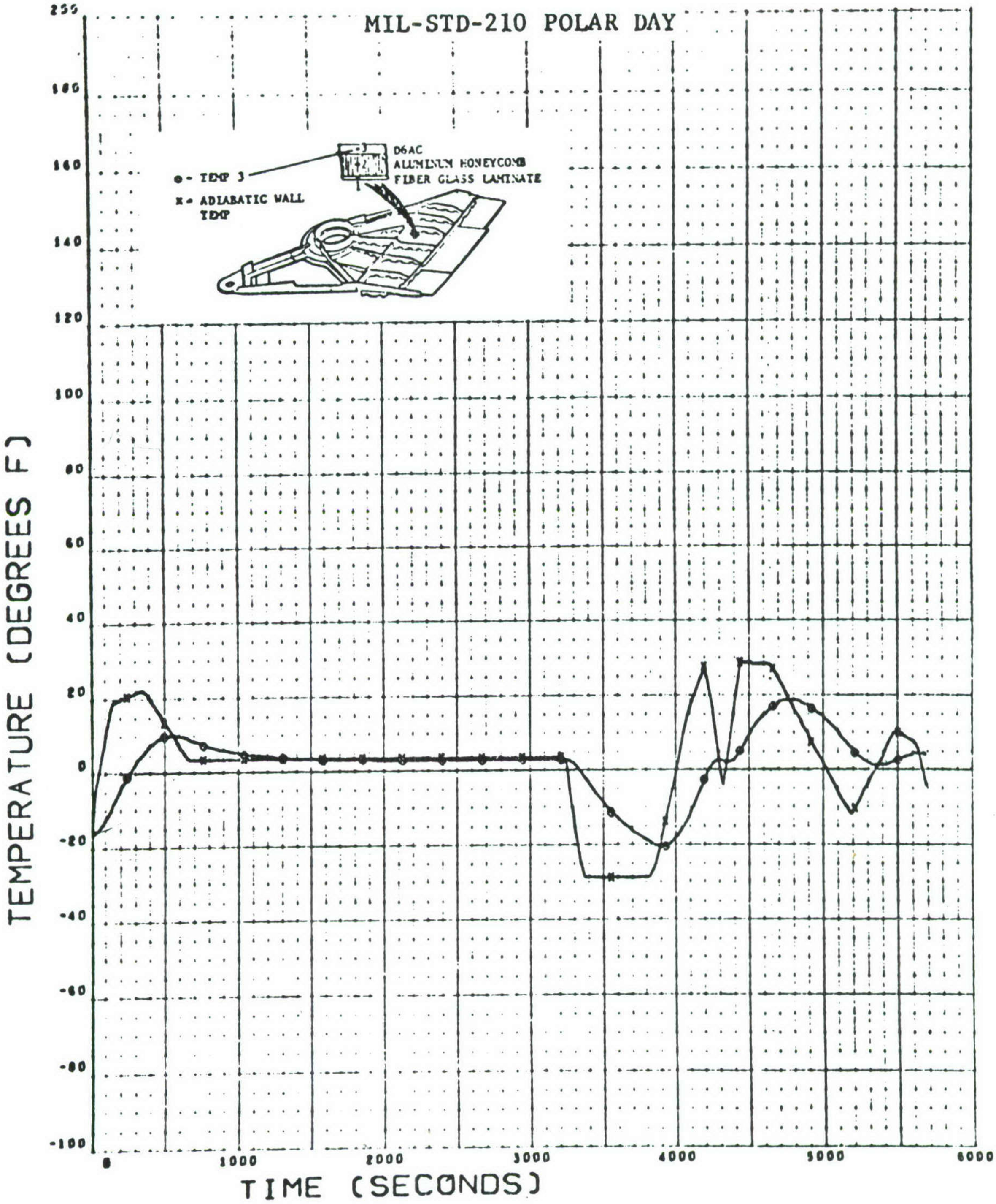
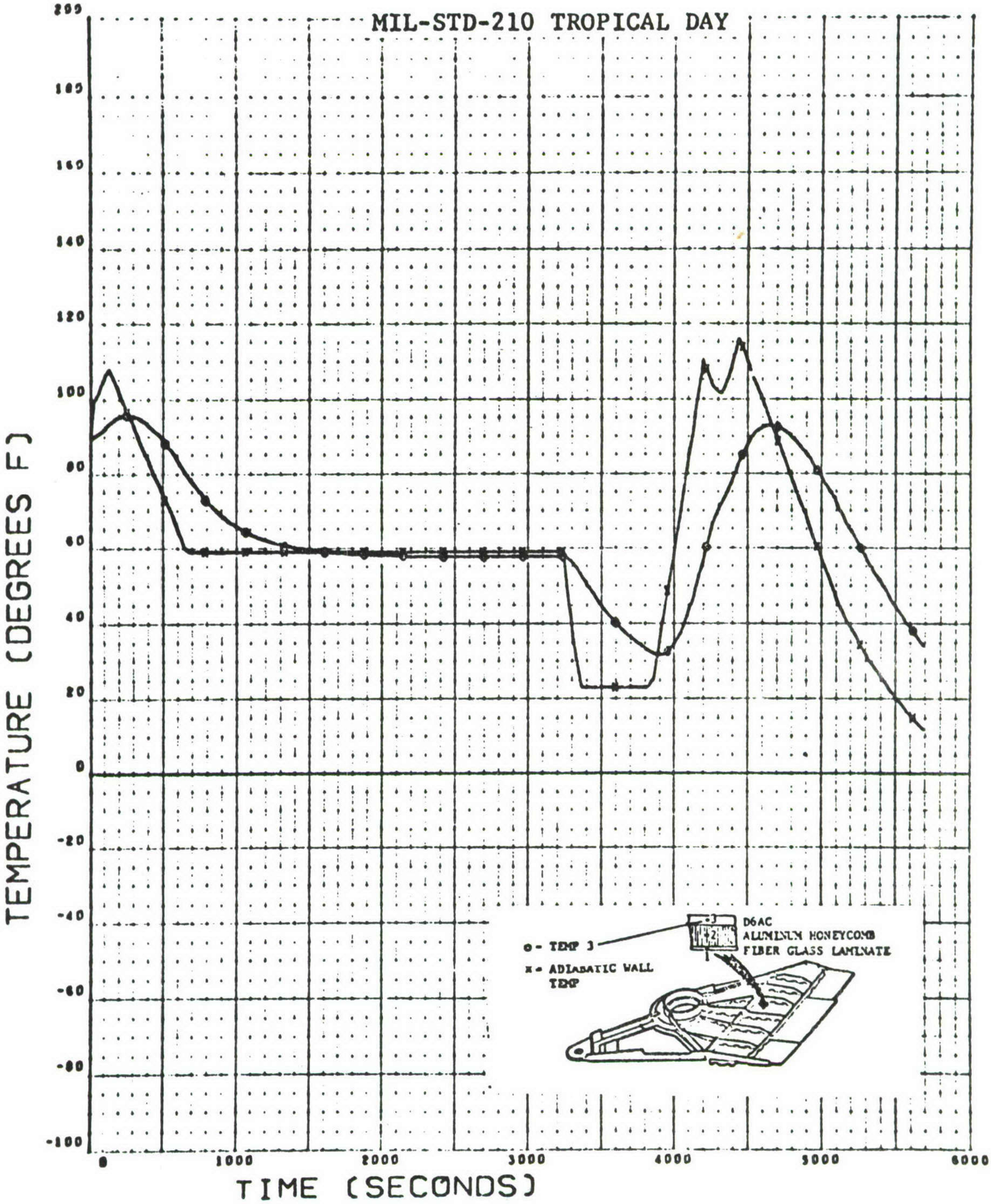


Figure 14

WING PIVOT FITTING 12W473
INBOARD OF SPLICE

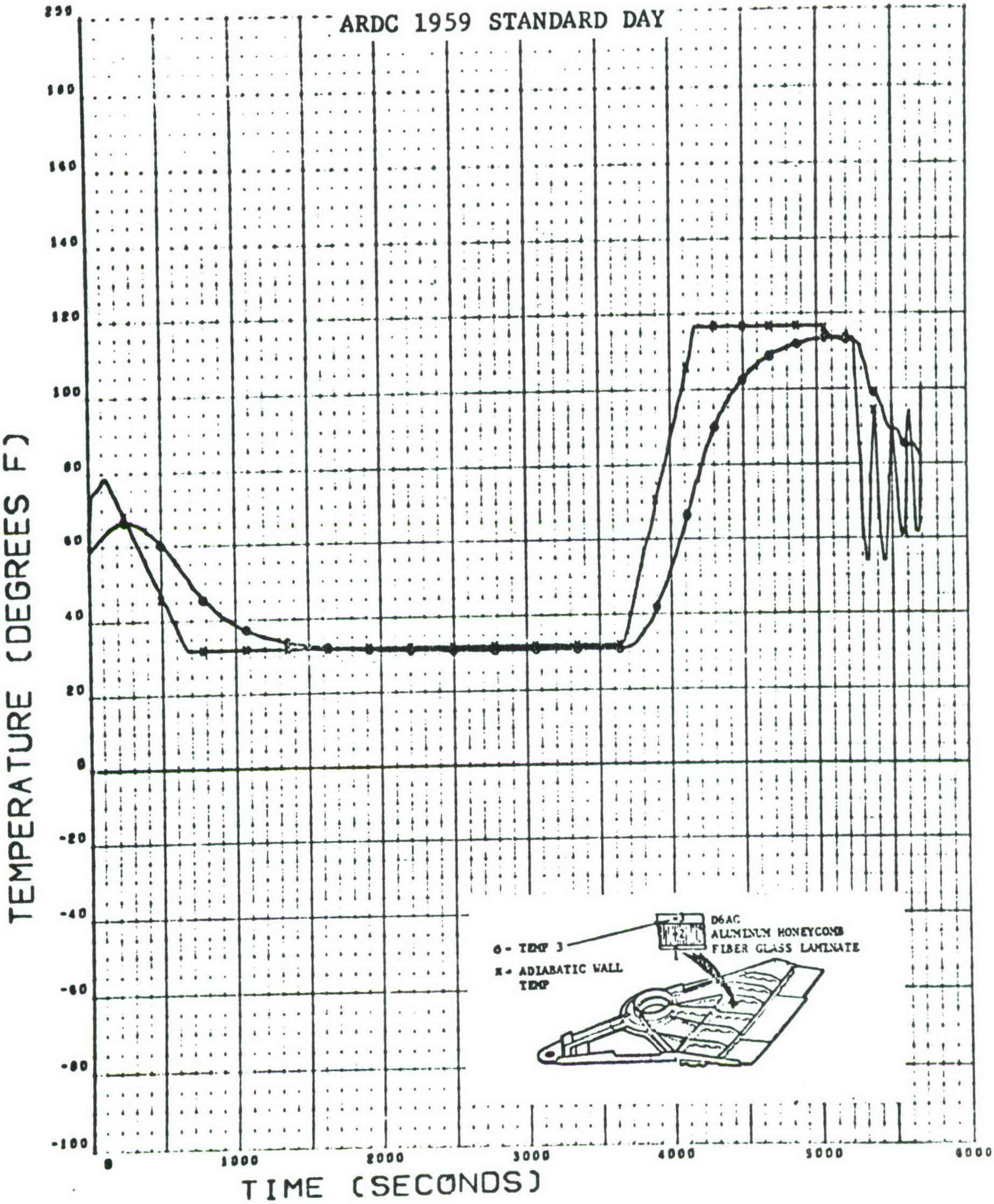
F-111 TRAINING MISSION TR(A)-2
(M = 1.5 DELETED)



WING PIVOT FITTING 12W473
INBOARD OF SPLICE

F-111 TRAINING MISSION TR(A)-5

ARDC 1959 STANDARD DAY



WING PIVOT FITTING 12W473
INBOARD OF SPLICE

F-111 TRAINING MISSION TR(A)-5

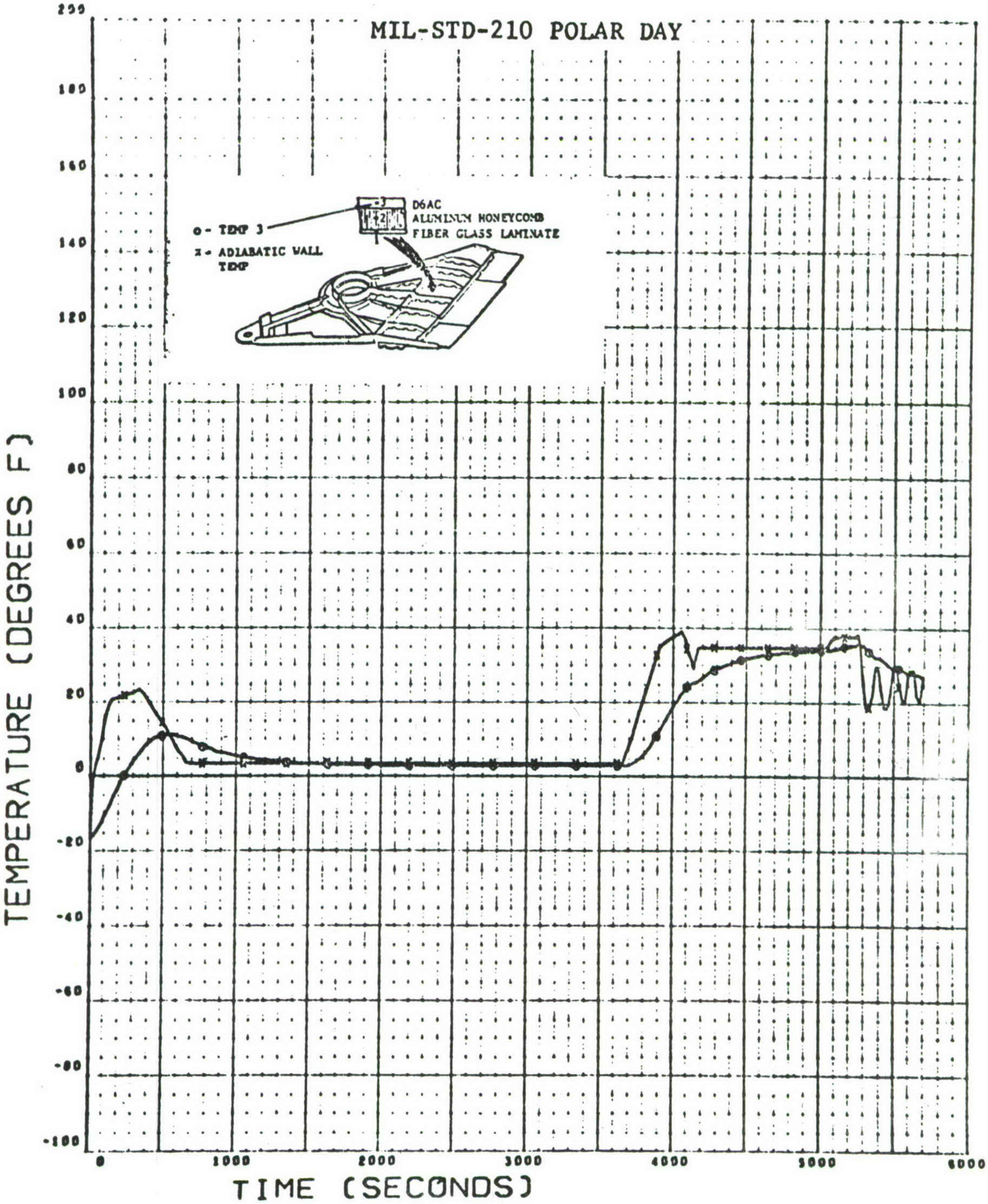
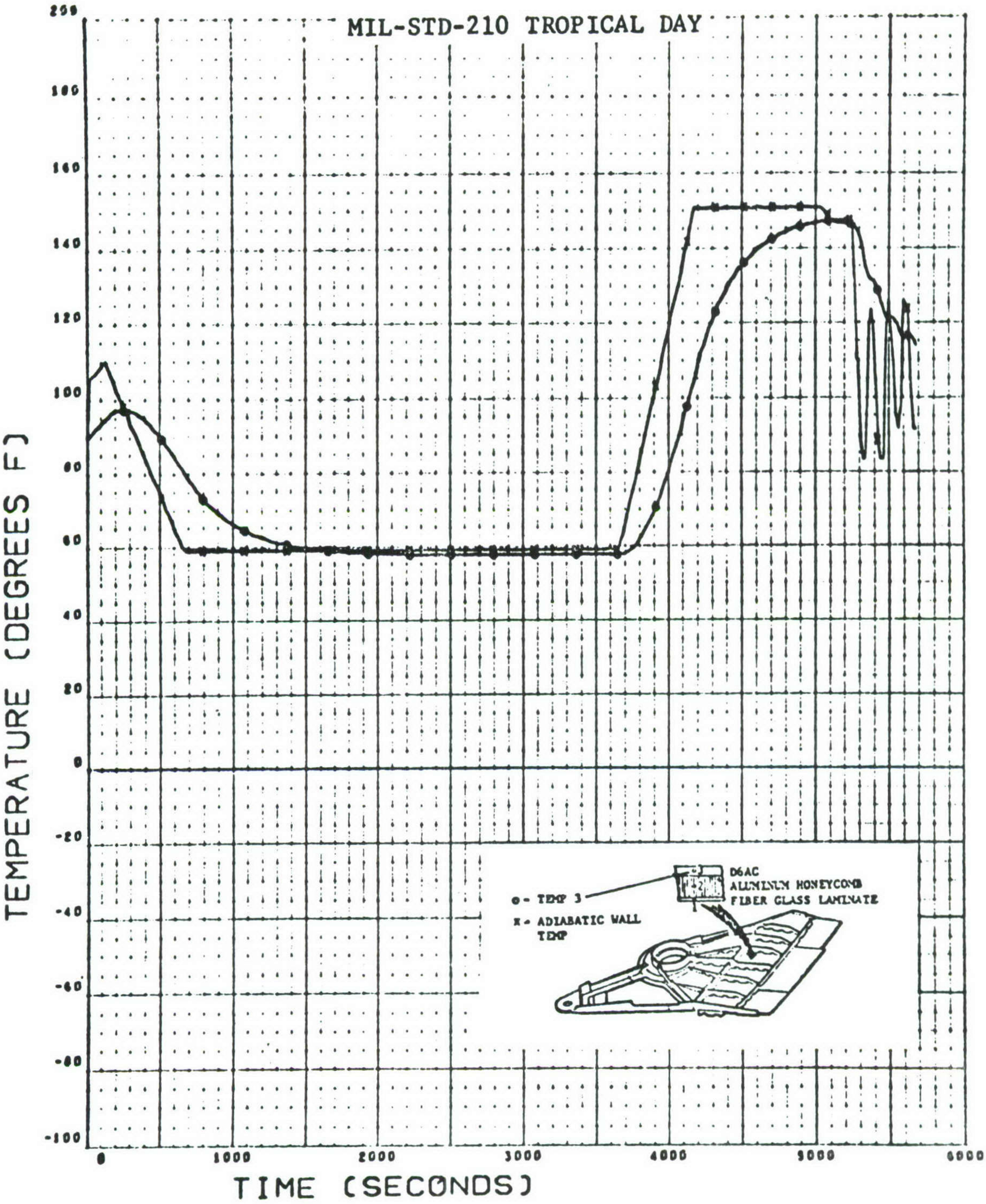


Figure 17

WING PIVOT FITTING 12W473
INBOARD OF SPLICE

F-111 TRAINING MISSION TR(A)-5



WING PIVOT FITTING 12W473
INBOARD OF SPLICE

F-111 20° DIVE BOMB RUN
AFTER 1 HOUR MAX LOITER

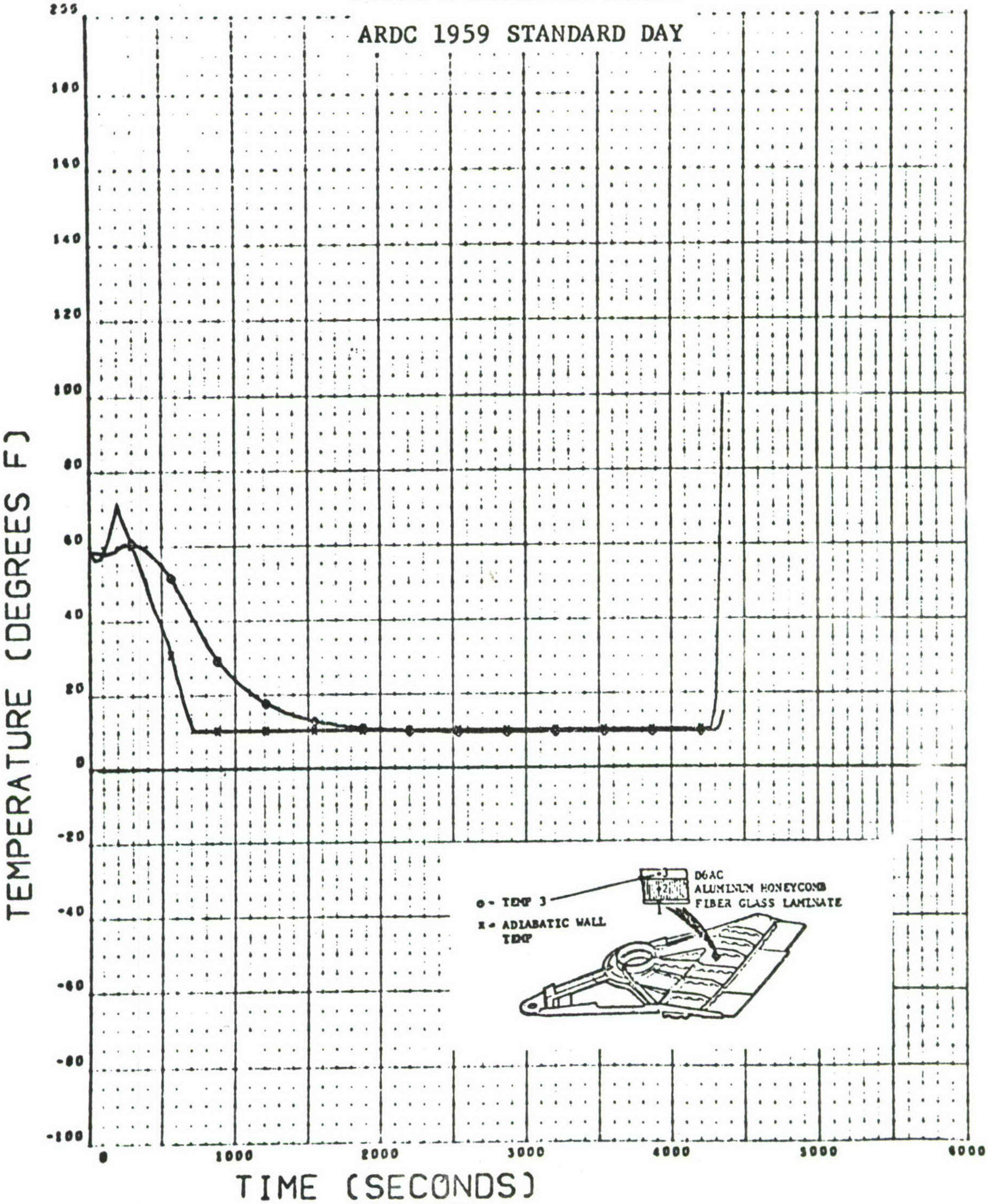


Figure 19

WING PIVOT FITTING 12W473
INBOARD OF SPLICE

F-111 20° DIVE BOMB RUN
AFTER 1 HOUR MAX LOITER

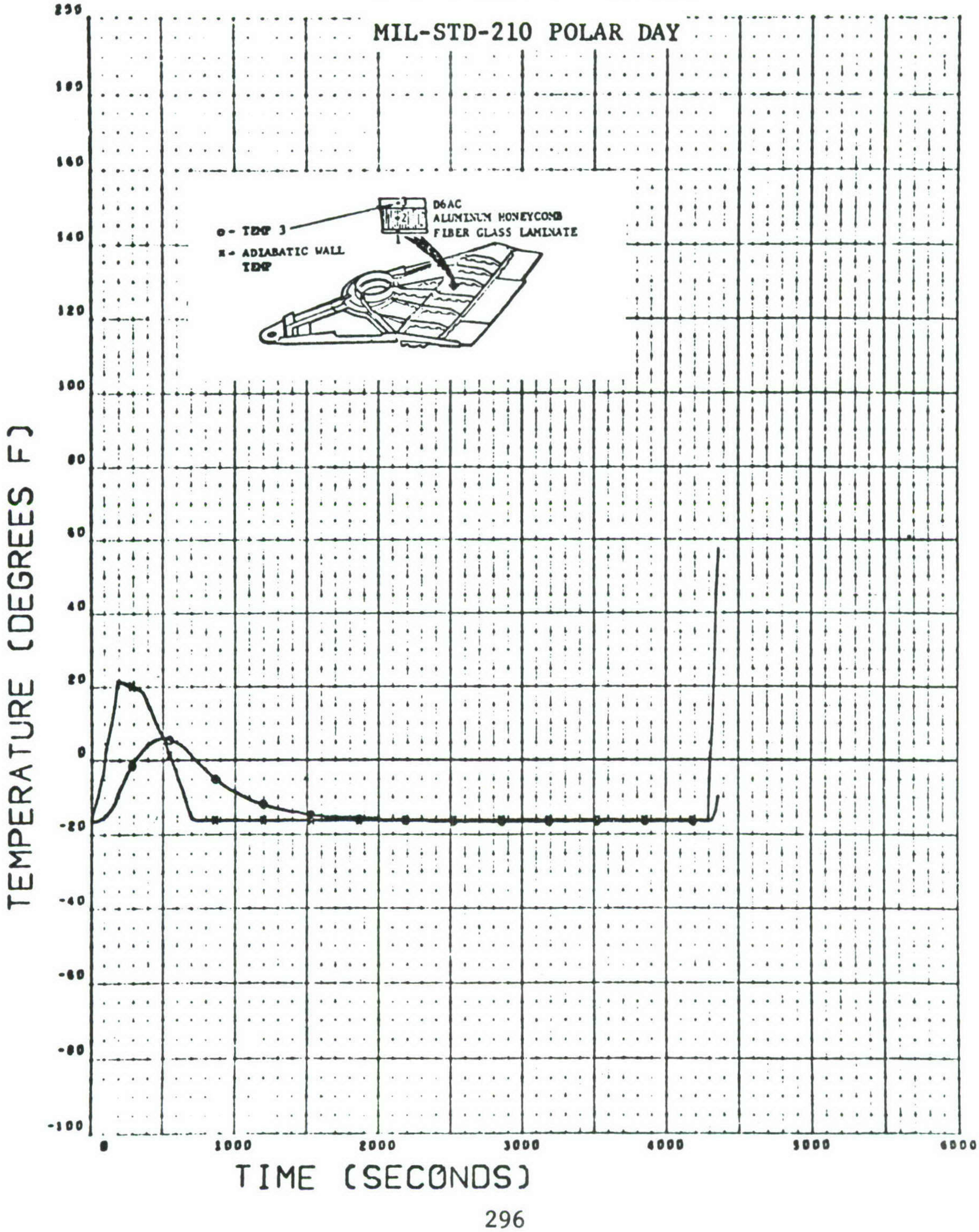


Figure 20

WING PIVOT FITTING 12W473
INBOARD OF SPLICE

F-111 20° DIVE BOMB RUN
AFTER 1 HOUR MAX LOITER

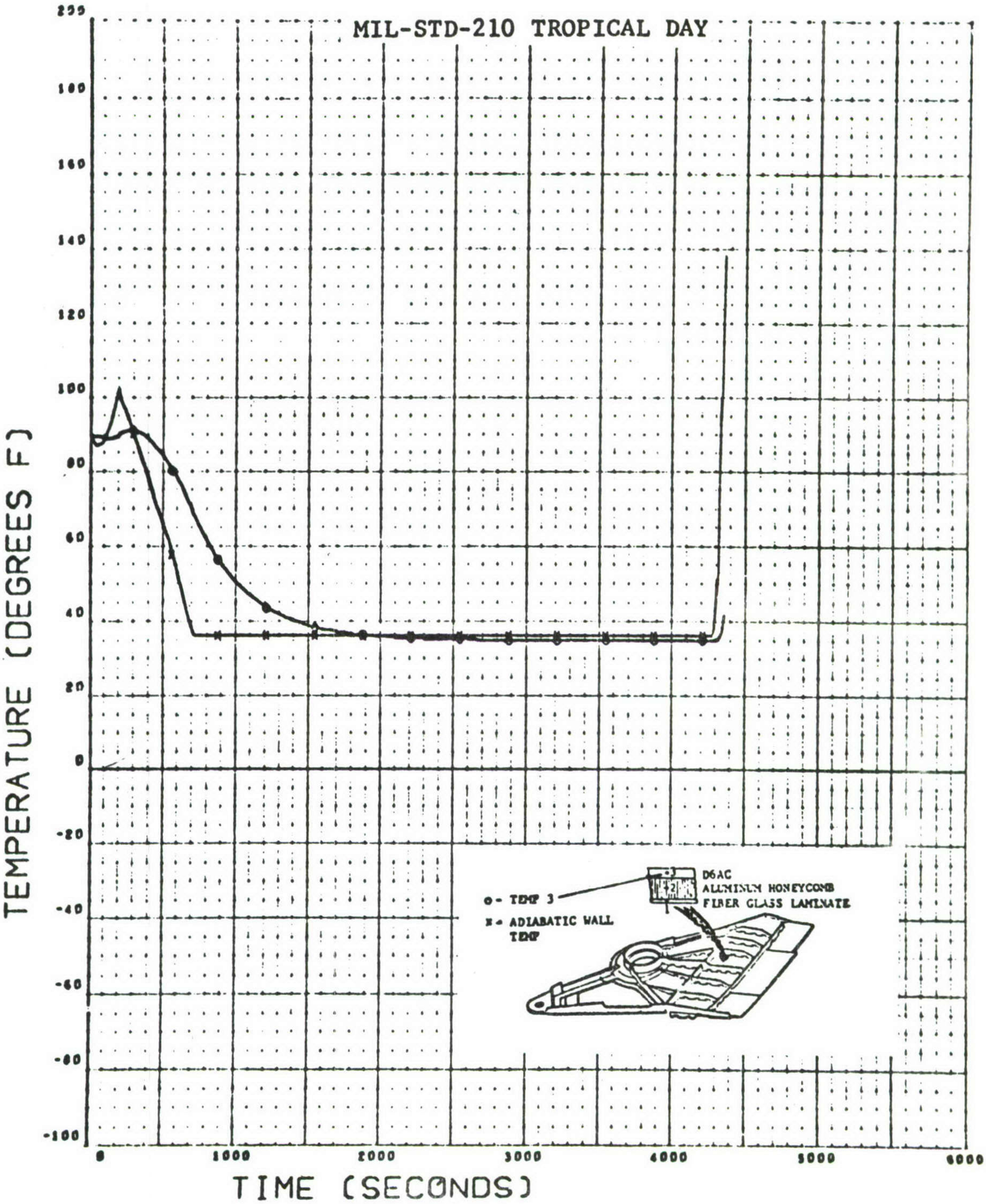


Figure 21

WING PIVOT FITTING 12W473
INBOARD OF SPLICE

F-111 FERRY MISSION

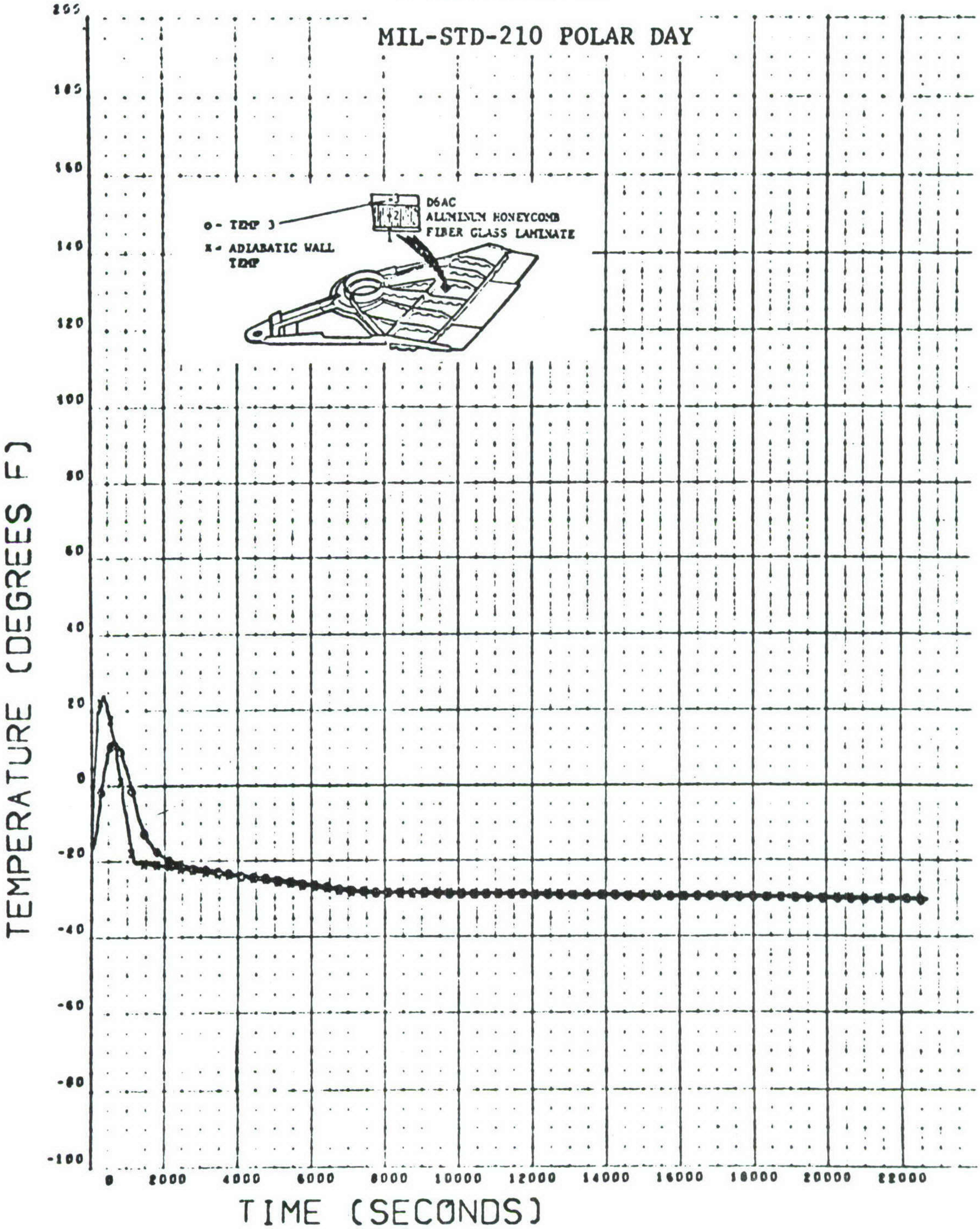


Figure 22

WING AND PIVOT FITTING COMBINED

F-111 TRAINING MISSION TR(A)-2
(M = 1.5 DELETED)

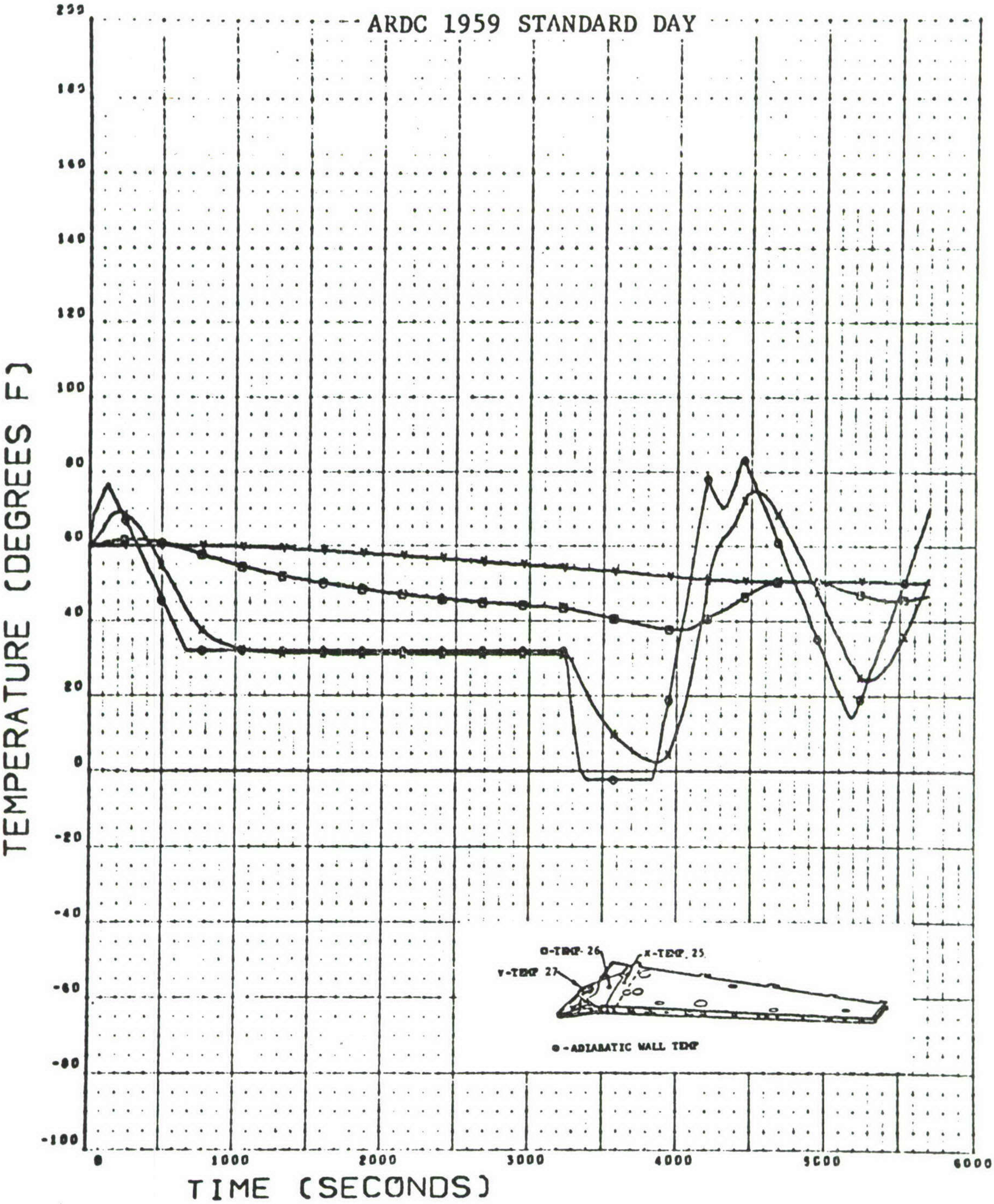


Figure 23

WING AND PIVOT FITTING COMBINED

F-111 TRAINING MISSION TR(A)-2
(M = 1.5 DELETED)

MIL-STD-210 POLAR DAY

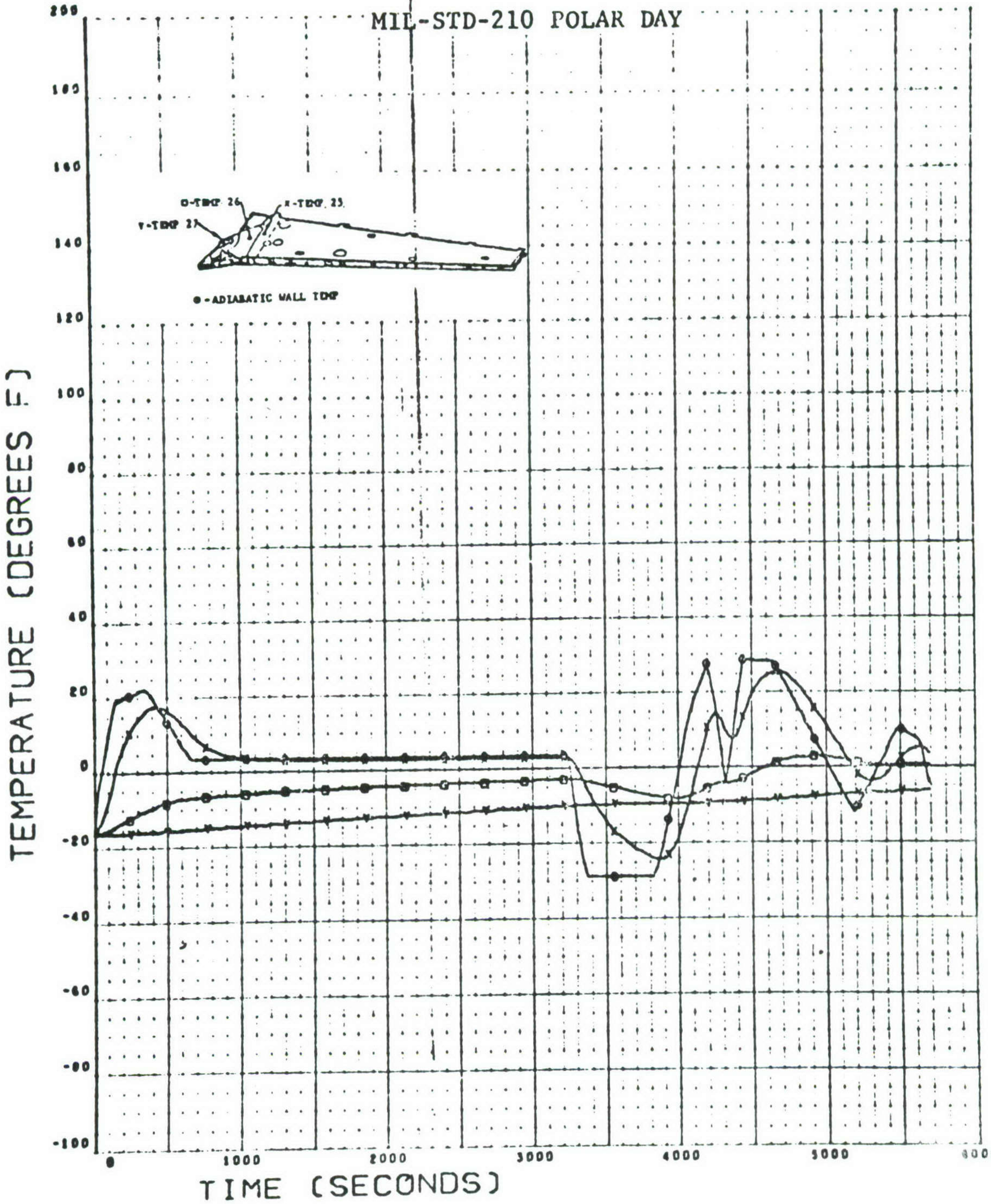


Figure 24

WING AND PIVOT FITTING COMBINED

F-111 TRAINING MISSION TR(A)-2
(M = 1.5 DELETED)

MIL-STD-210 TROPICAL DAY

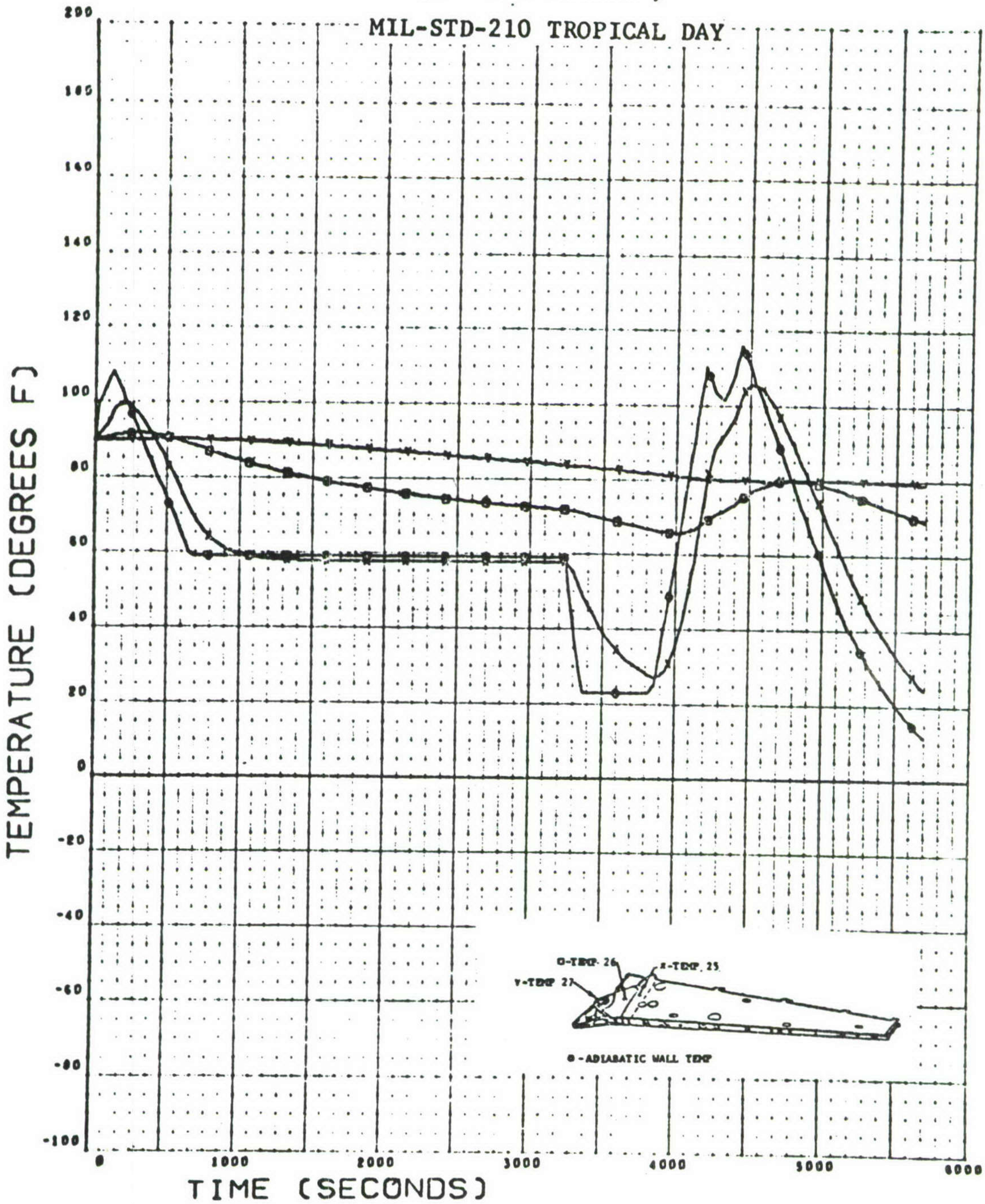


Figure 25

WING AND PIVOT FITTING COMBINED

F-111 TRAINING MISSION TR(A)-5

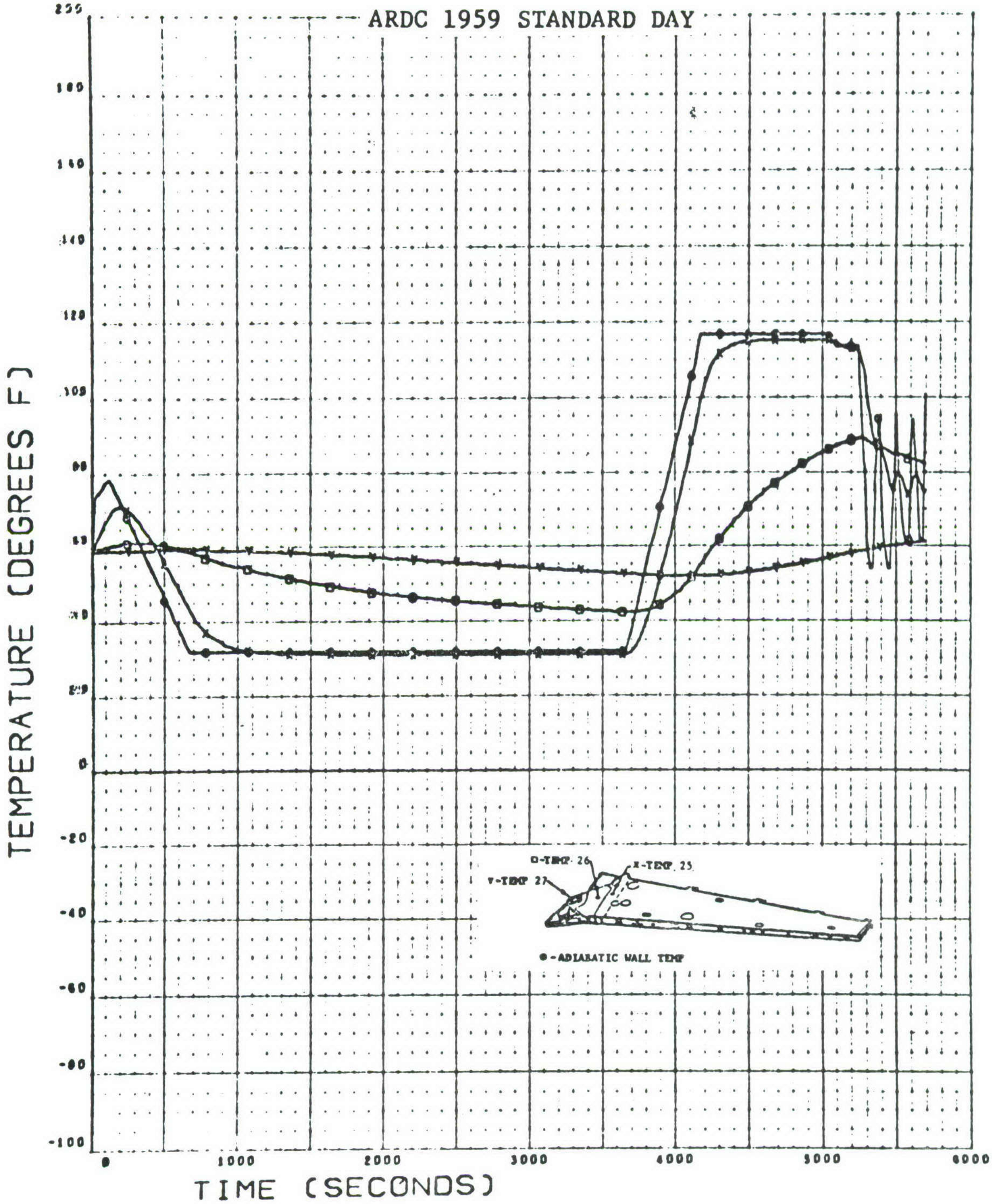
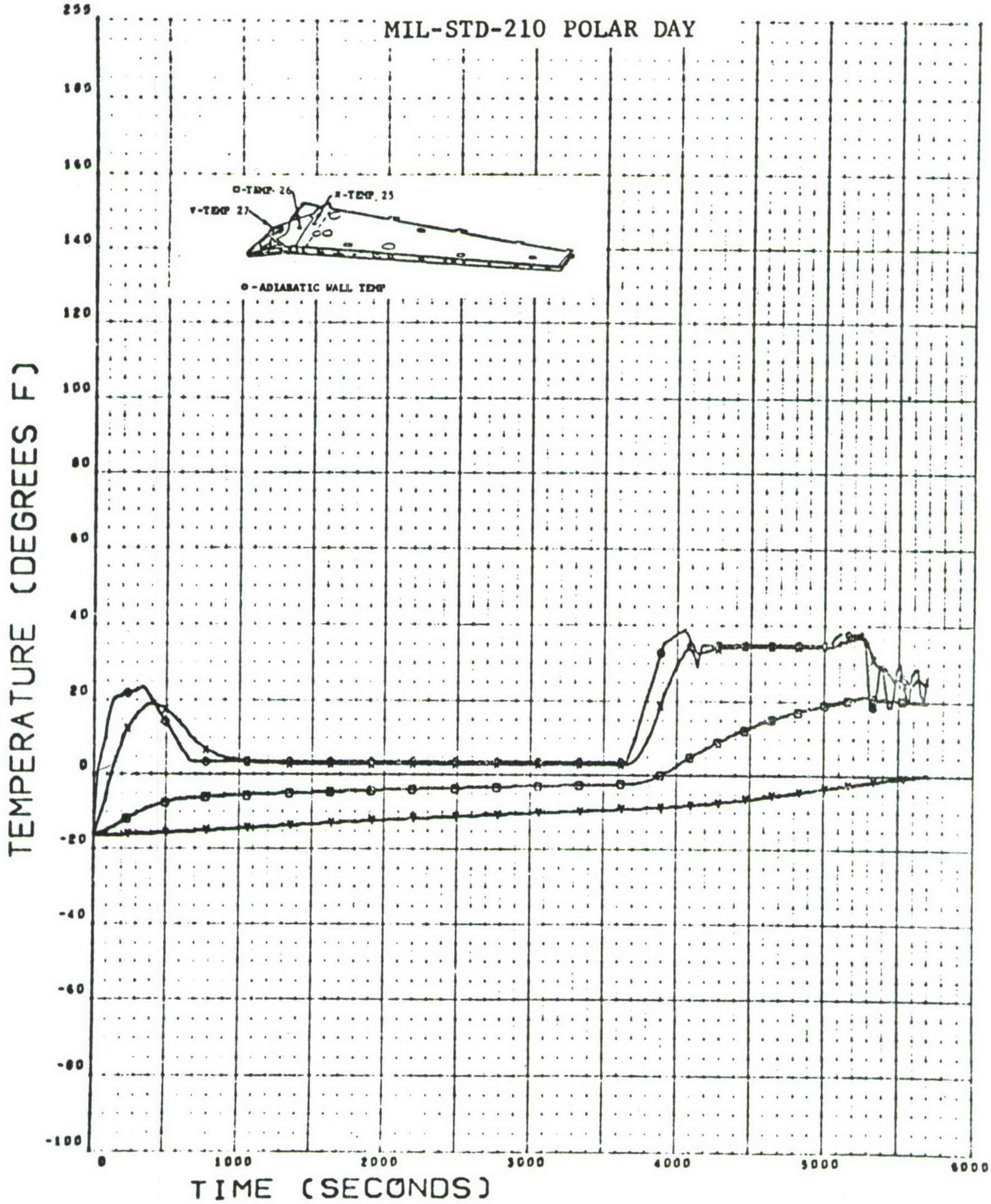


Figure 26

WING AND PIVOT FITTING COMBINED

F-111 TRAINING MISSION TR(A)-5



WING AND PIVOT FITTING COMBINED

F-111 TRAINING MISSION TR(A)-5

MIL-STD-210 TROPICAL DAY

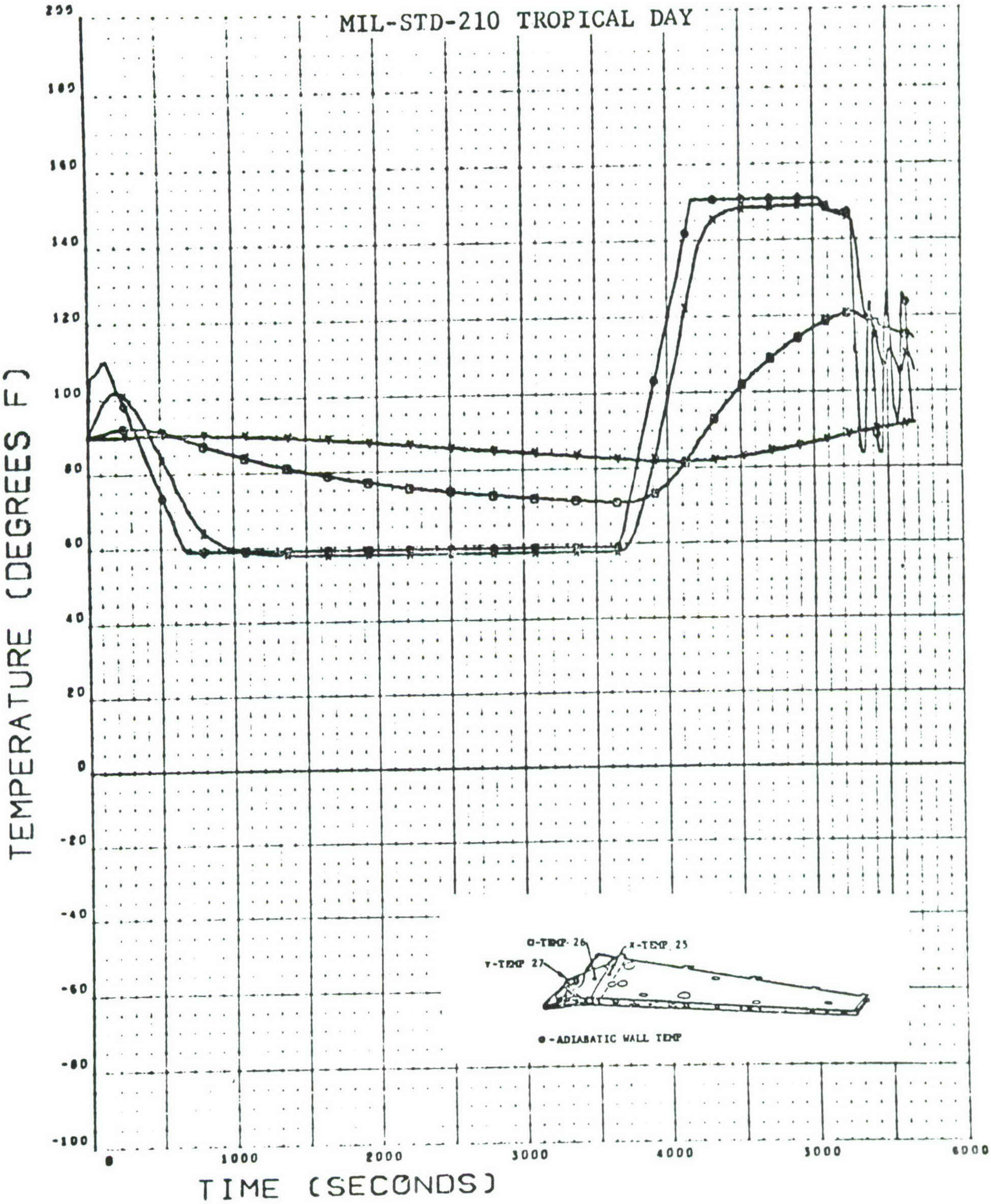
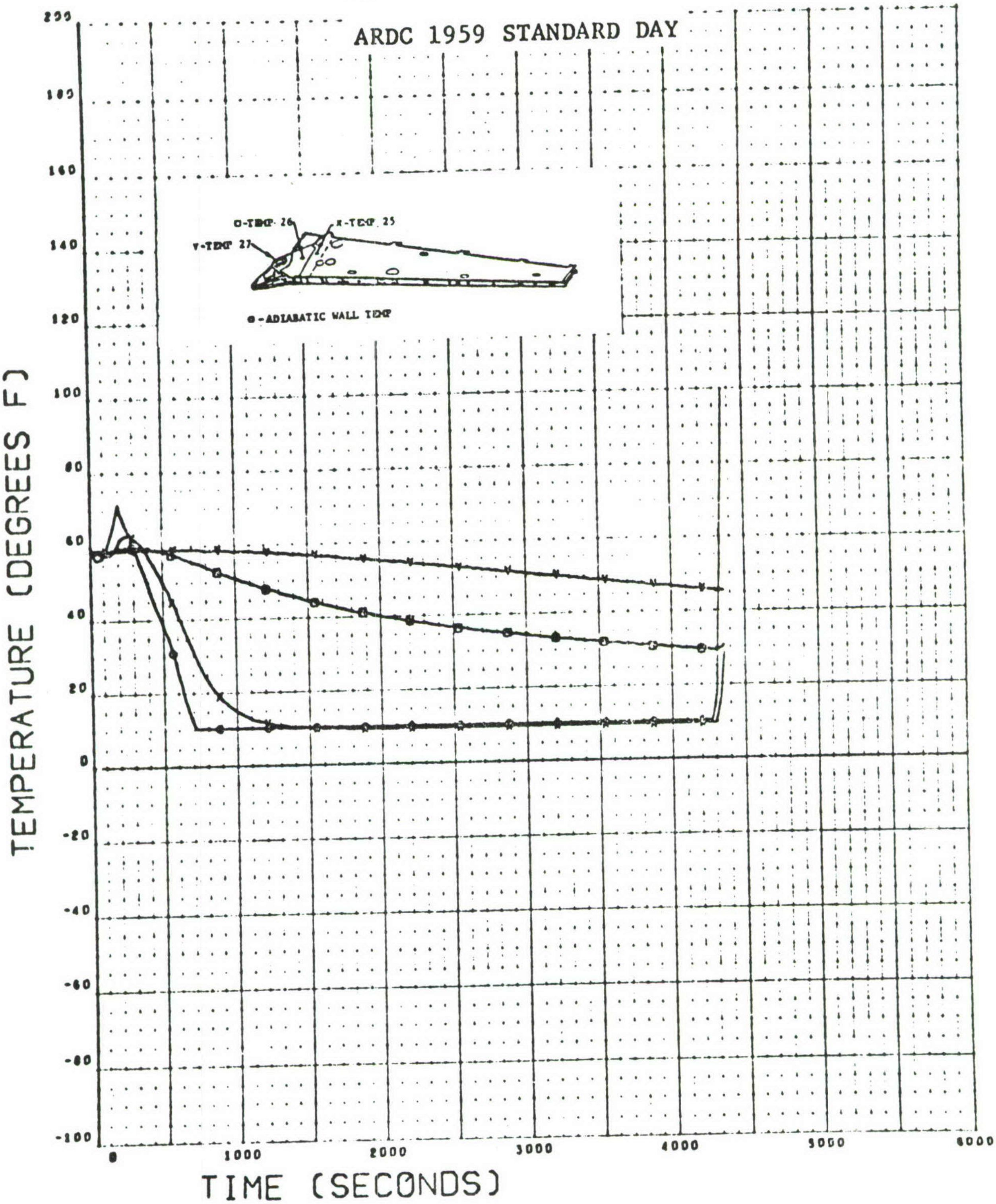


Figure 28

WING AND PIVOT FITTING COMBINED

F-111 20° DIVE BOMB RUN
AFTER 1 HOUR MAX LOITER

ARDC 1959 STANDARD DAY



WING AND PIVOT FITTING COMBINED

F-111 20^{o} DIVE BOMB RUN
AFTER 1 HOUR MAX LOITER

MIL-STD-210 POLAR DAY

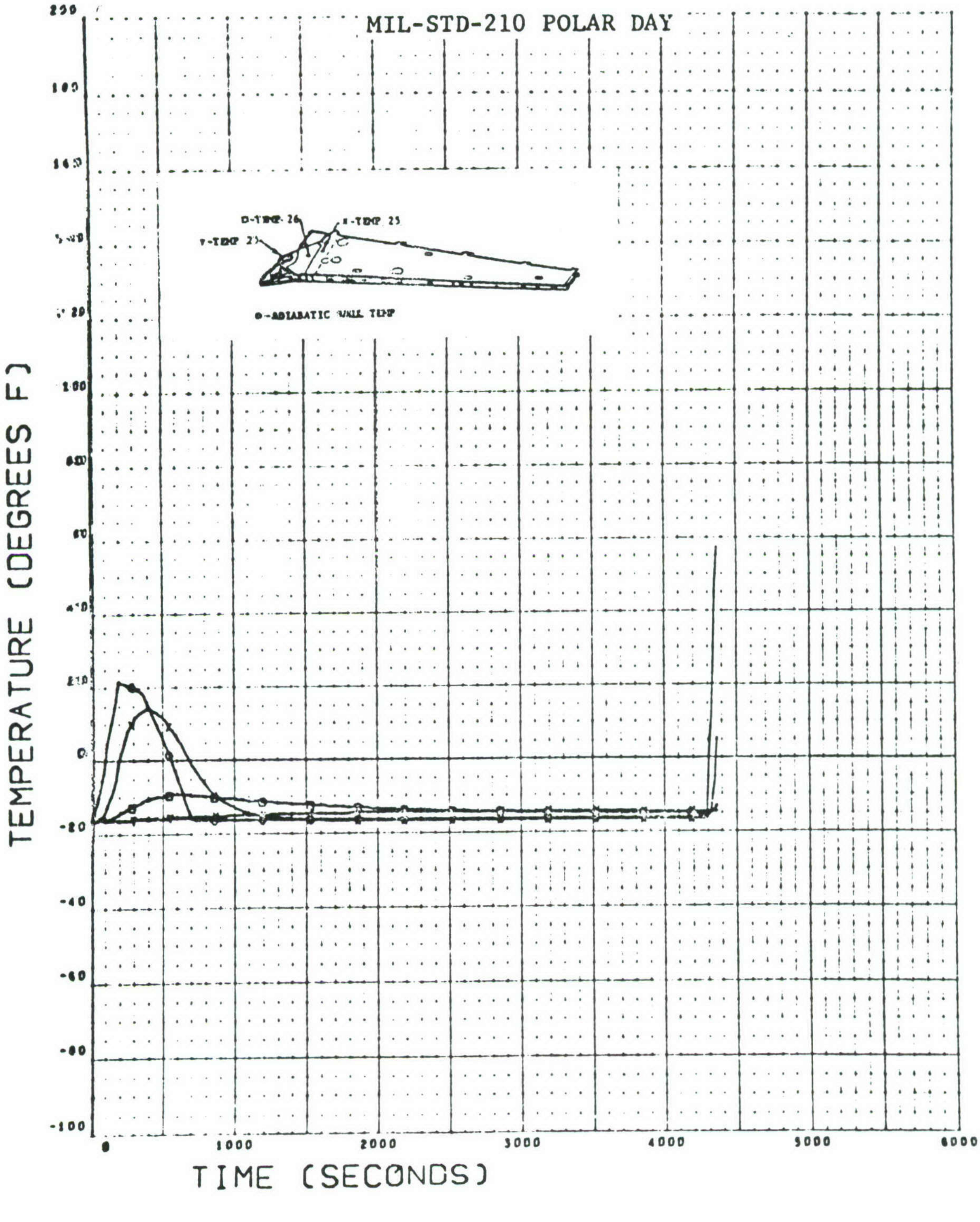


Figure 30

WING AND PIVOT FITTING COMBINED

F-111 20° DIVE BOMB RUN
AFTER 1 HOUR MAX LOITER

MIL-STD-210 TROPICAL DAY

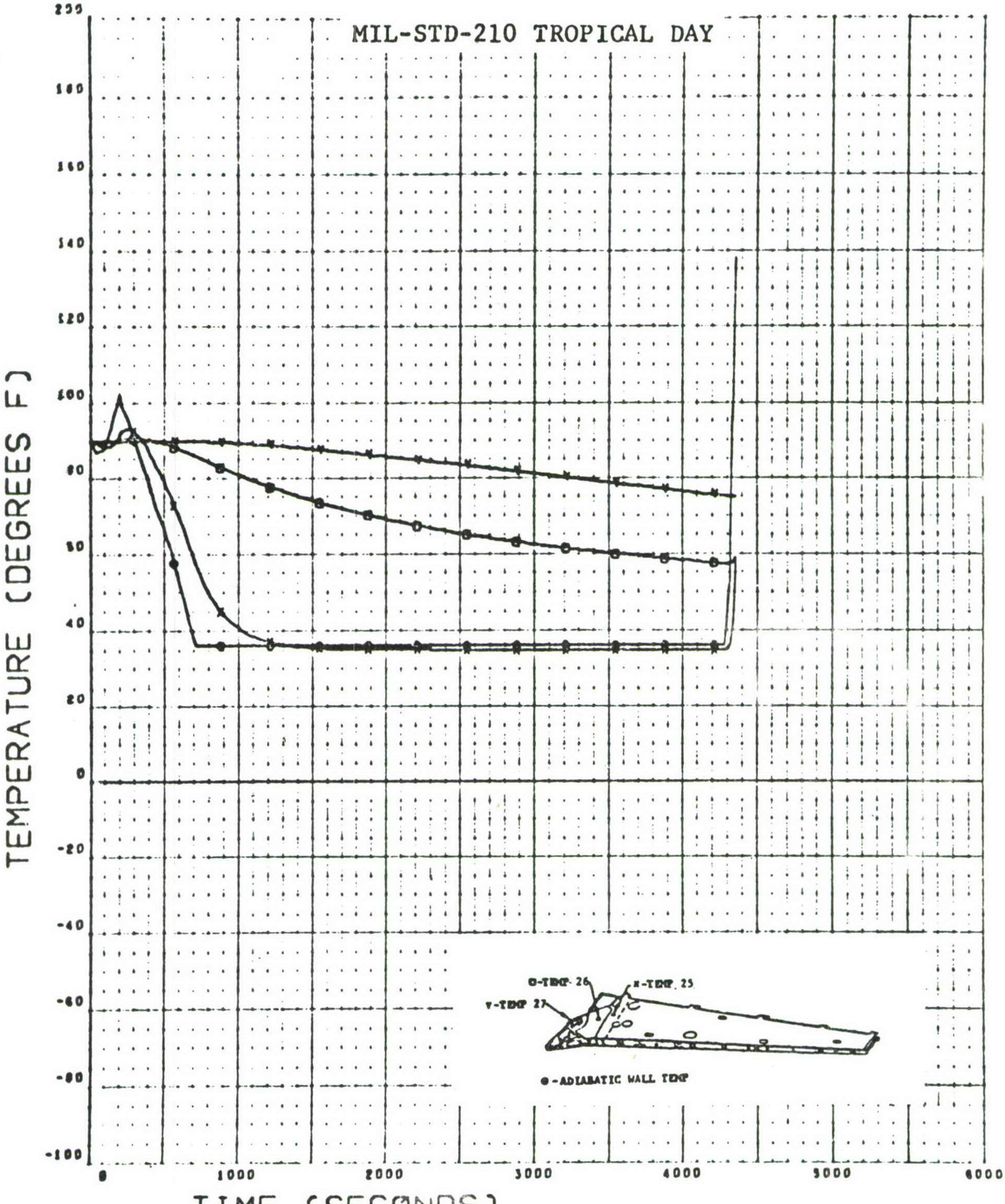


Figure 31

WING AND PIVOT FITTING COMBINED

F-111 FERRY MISSION

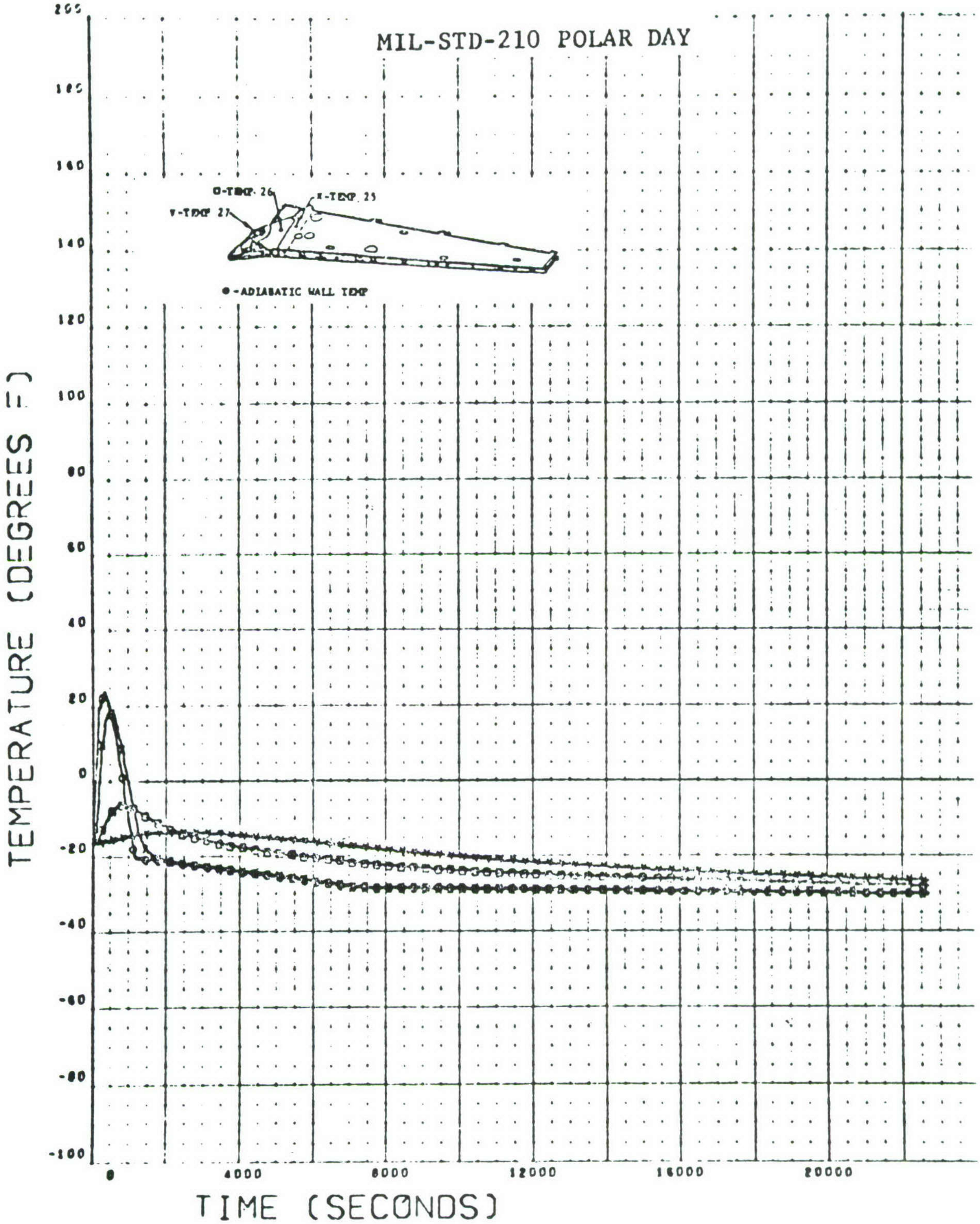


Figure 32

F-111 TRAINING MISSION TR(A)-2

M = 1.5 DELETED

FSG. STA. 496 BULKHEAD
12B2910

ARDC 1959 STANDARD DAY

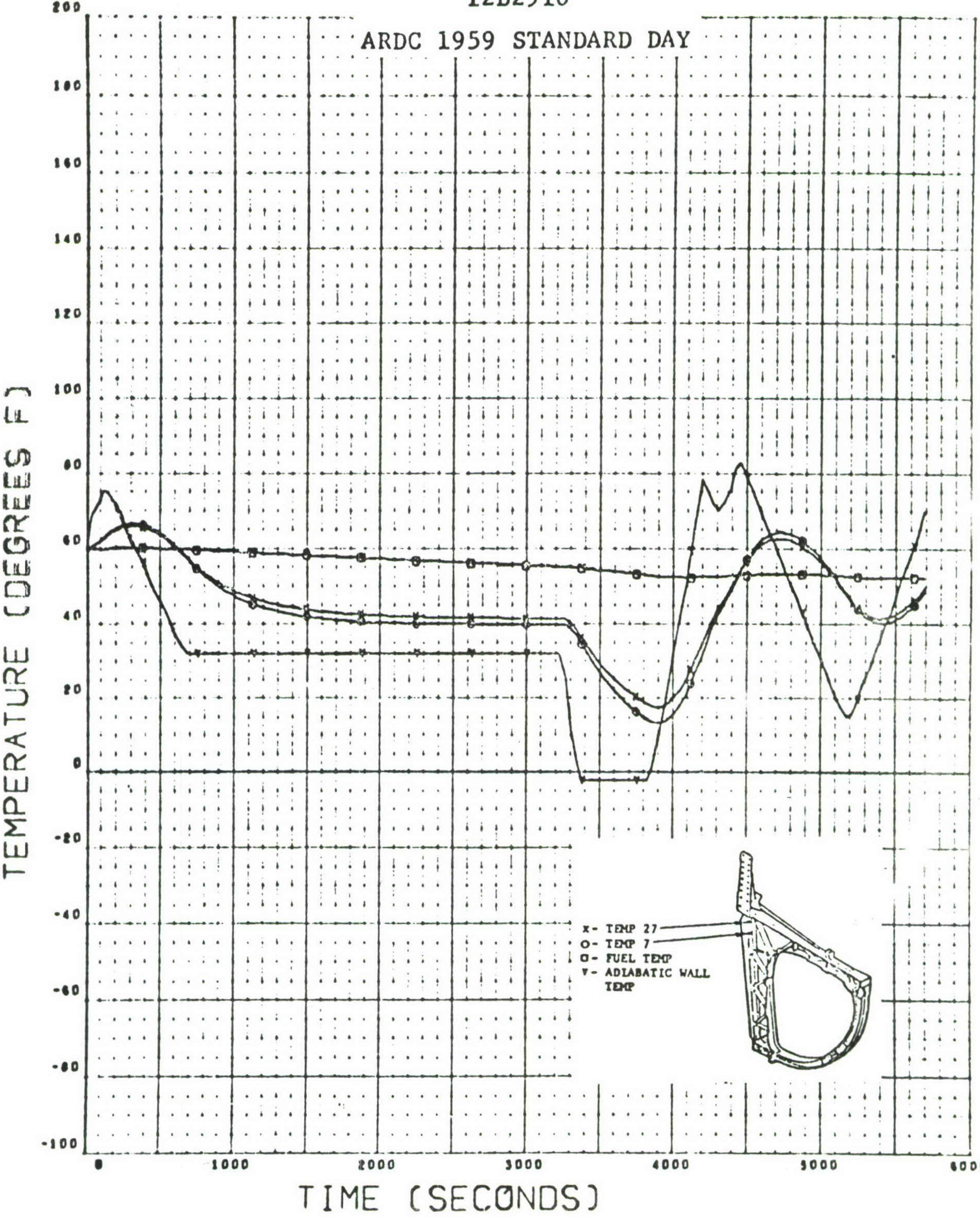


Figure 33

FSG. STA. 496 BULKHEAD
12B2910

F-111 TRAINING MISSION TR(A)-2
(M = 1.5 DELETED)

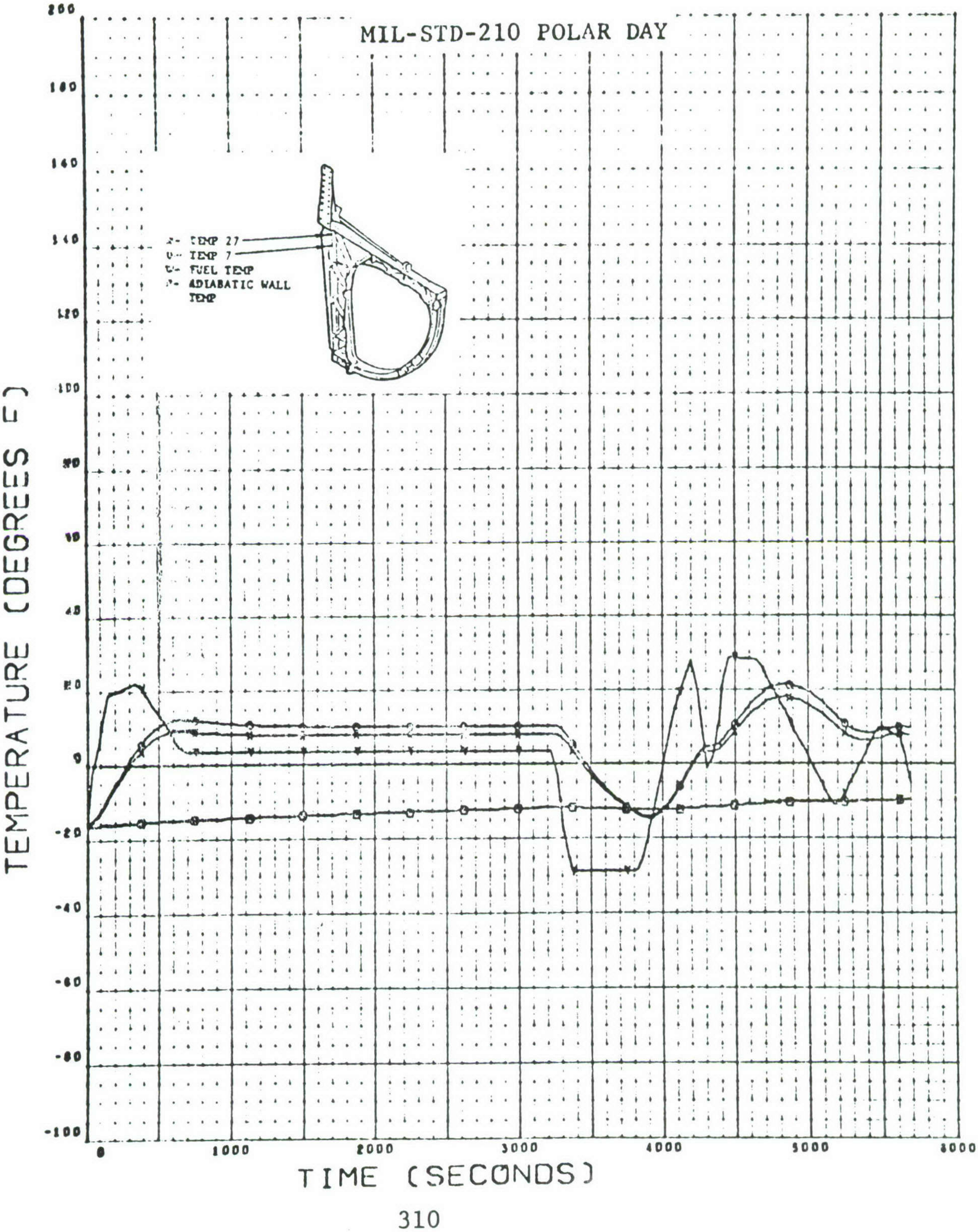


Figure 34

FSG. STA. 496 BULKHEAD
12B2910

F-111 TRAINING MISSION TR(A)-2
(M = 1.5 DELETED)

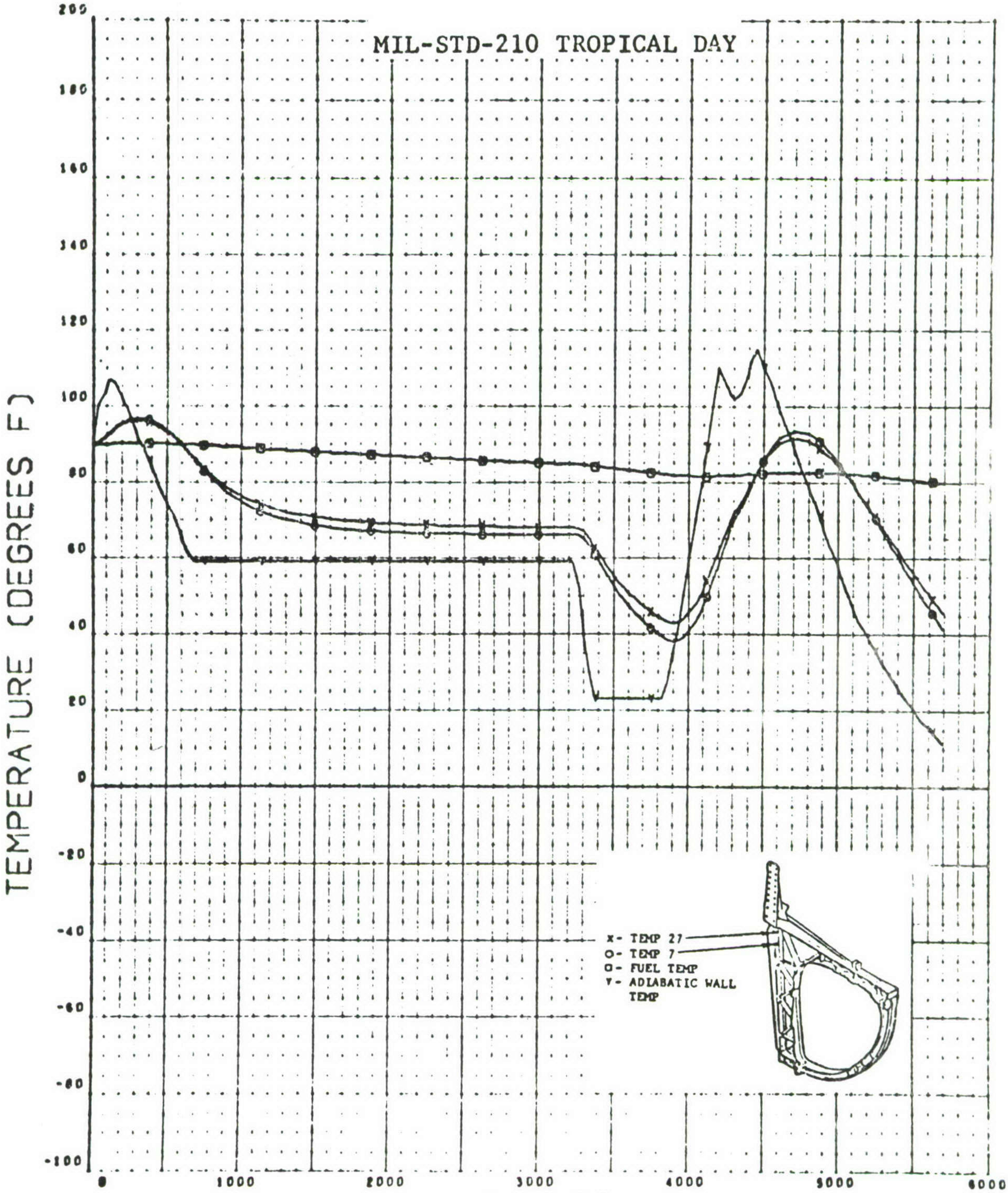


Figure 35

F-111 TRAINING MISSION TR(A)-5

FSG. STA. 496 BULKHEAD
12B2910

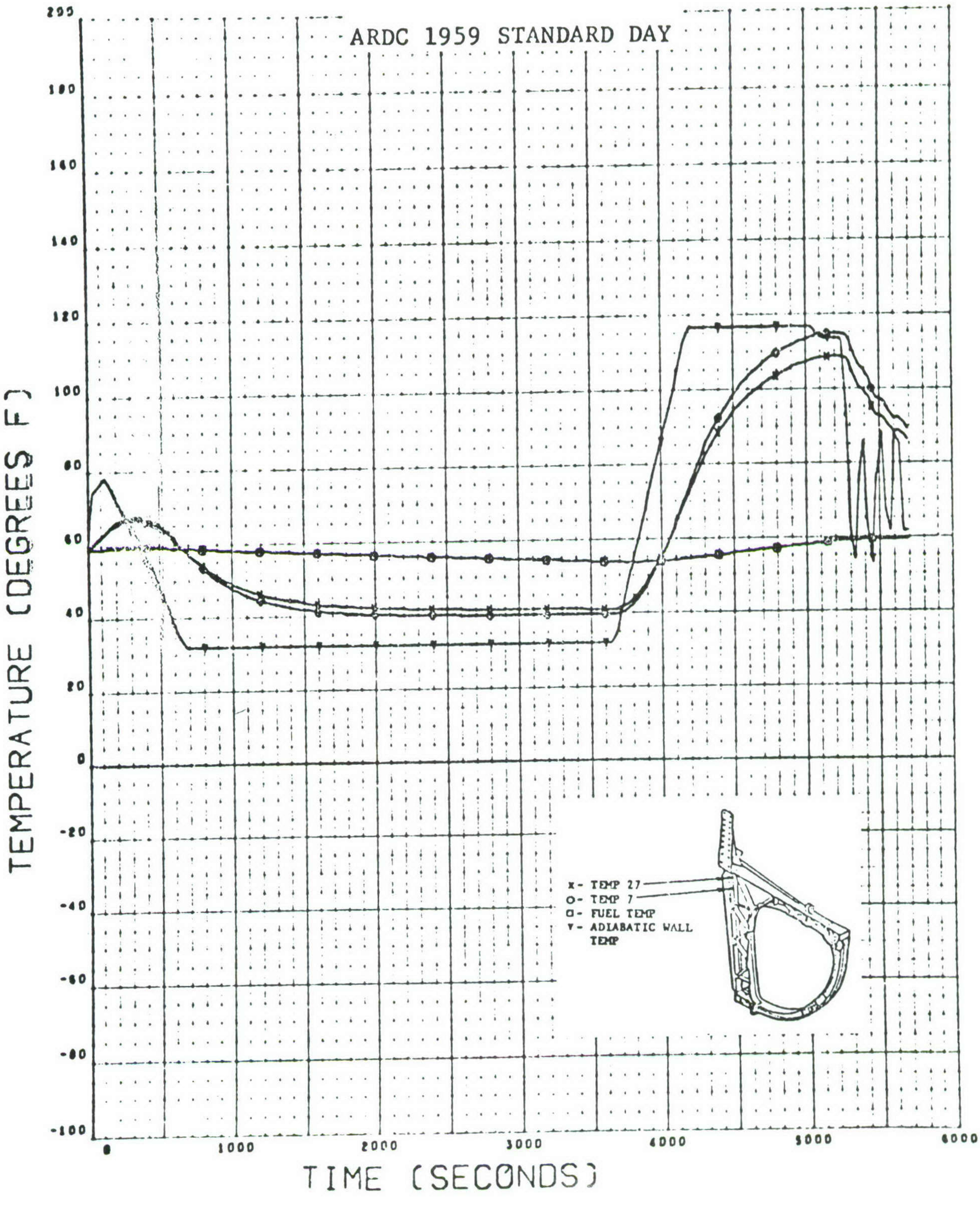


Figure 36

F-111 TRAINING MISSION TR(A)-5

FSG. STA. 496 BULKHEAD
12B2910

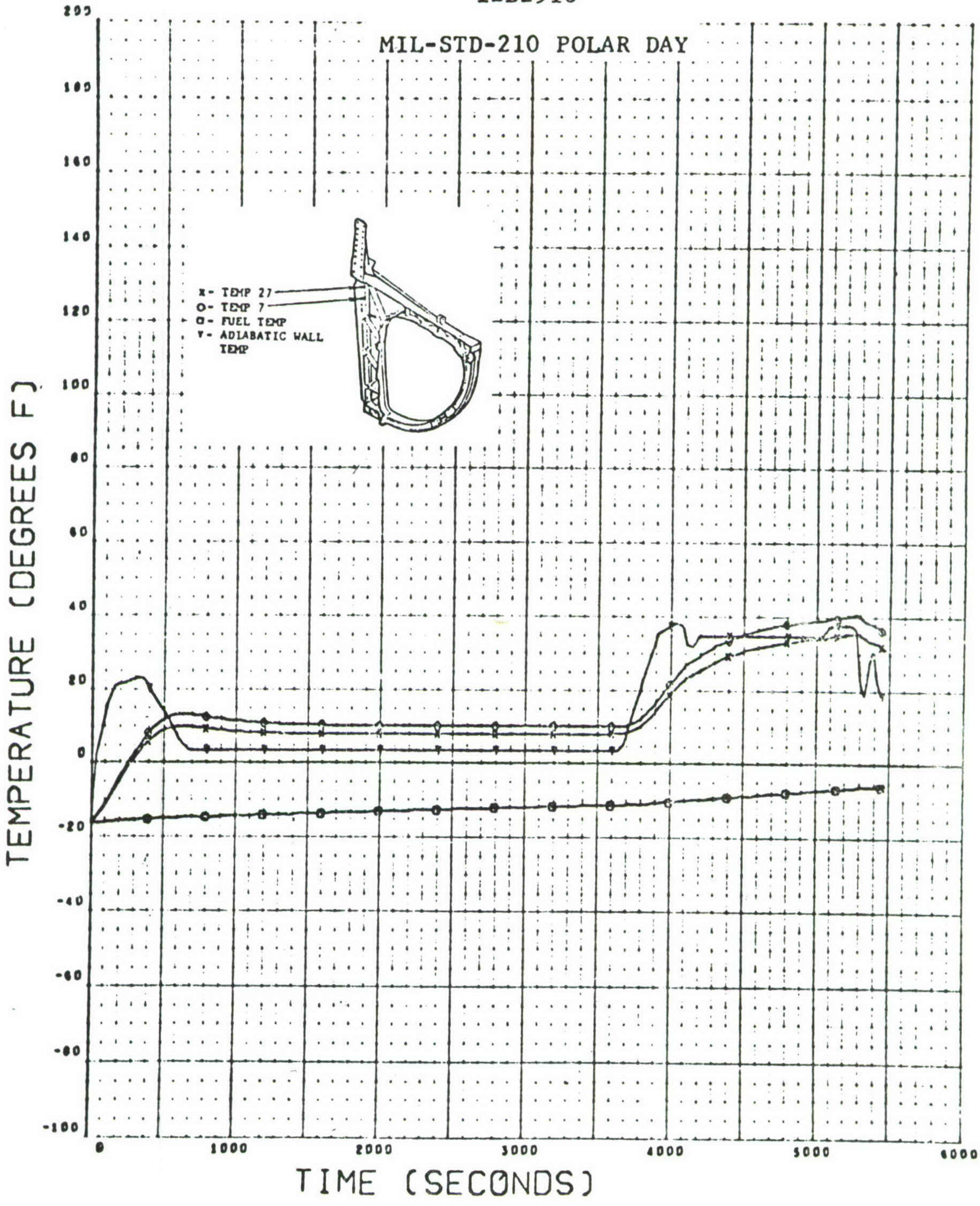


Figure 37

F-111 TRAINING MISSION TR(A)-5

FSG. STA. 496 BULKHEAD
12B2910

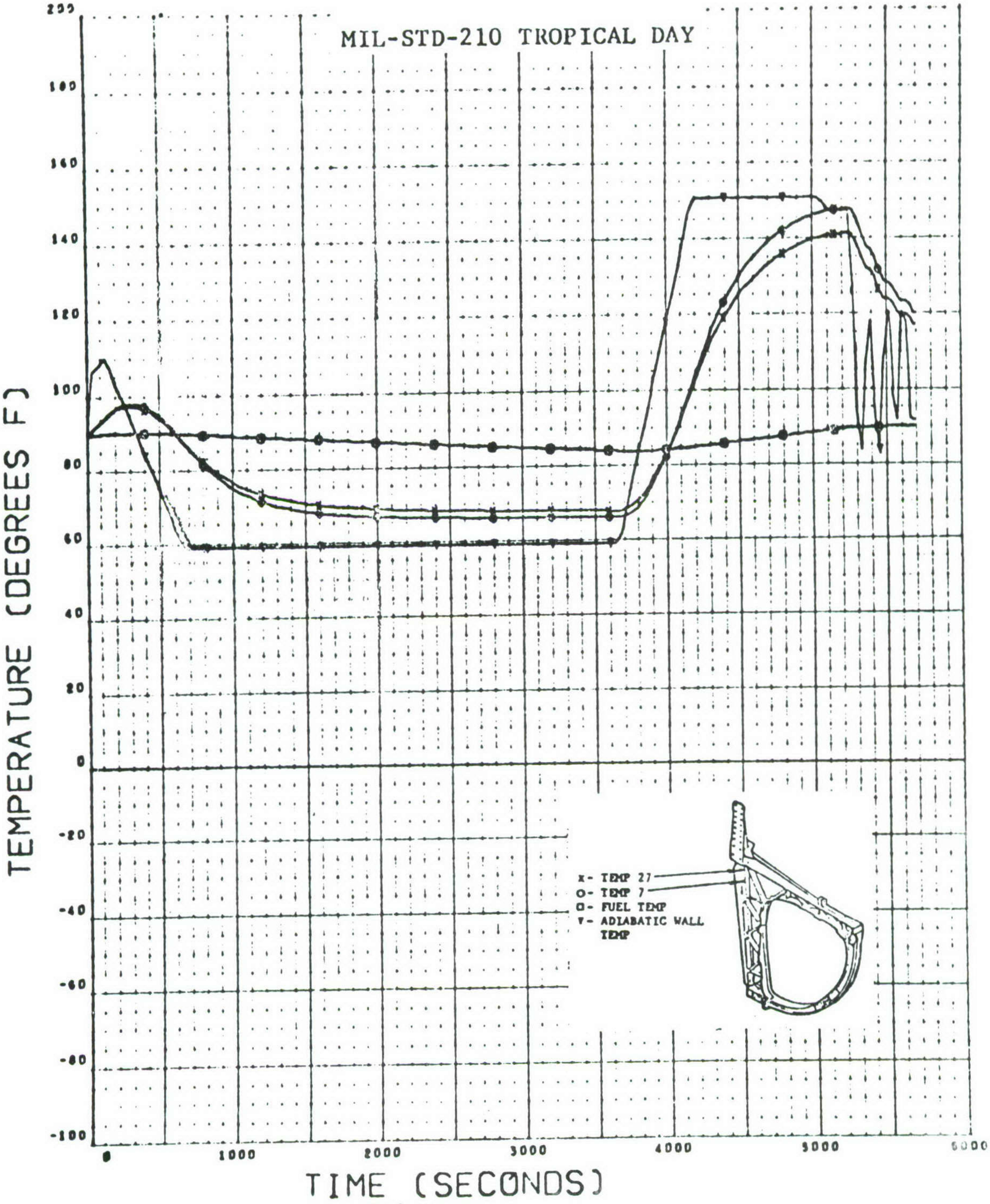


Figure 38

F-111 20° DIVE BOMB RUN
AFTER 1 HOUR MAX LOITER

FSG. STA. 496 BULKHEAD
12B2910

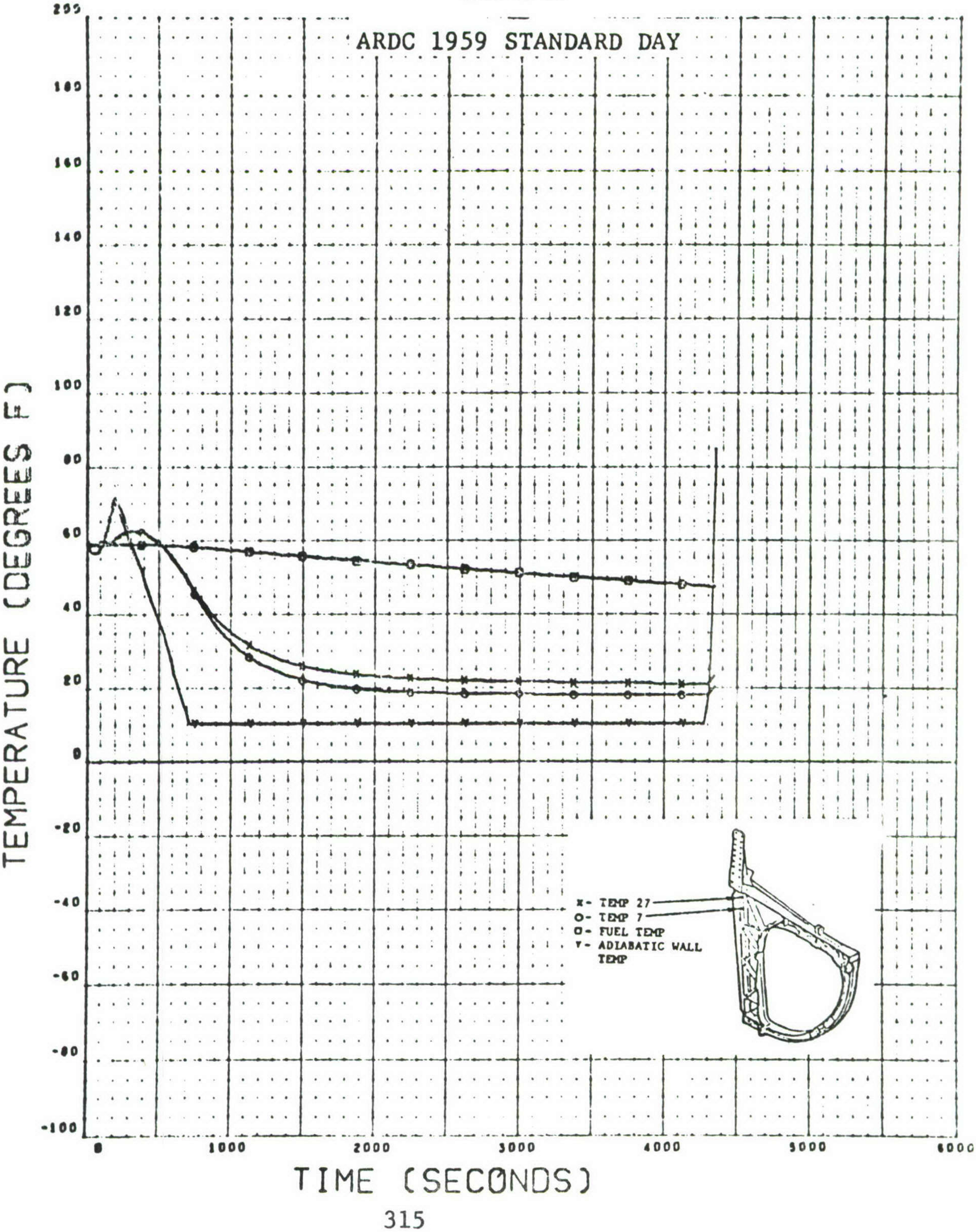


Figure 39

F-111 20° DIVE BOMB RUN
AFTER 1 HOUR MAX LOITER

FSG. STA. 496 BULKHEAD
12B2910

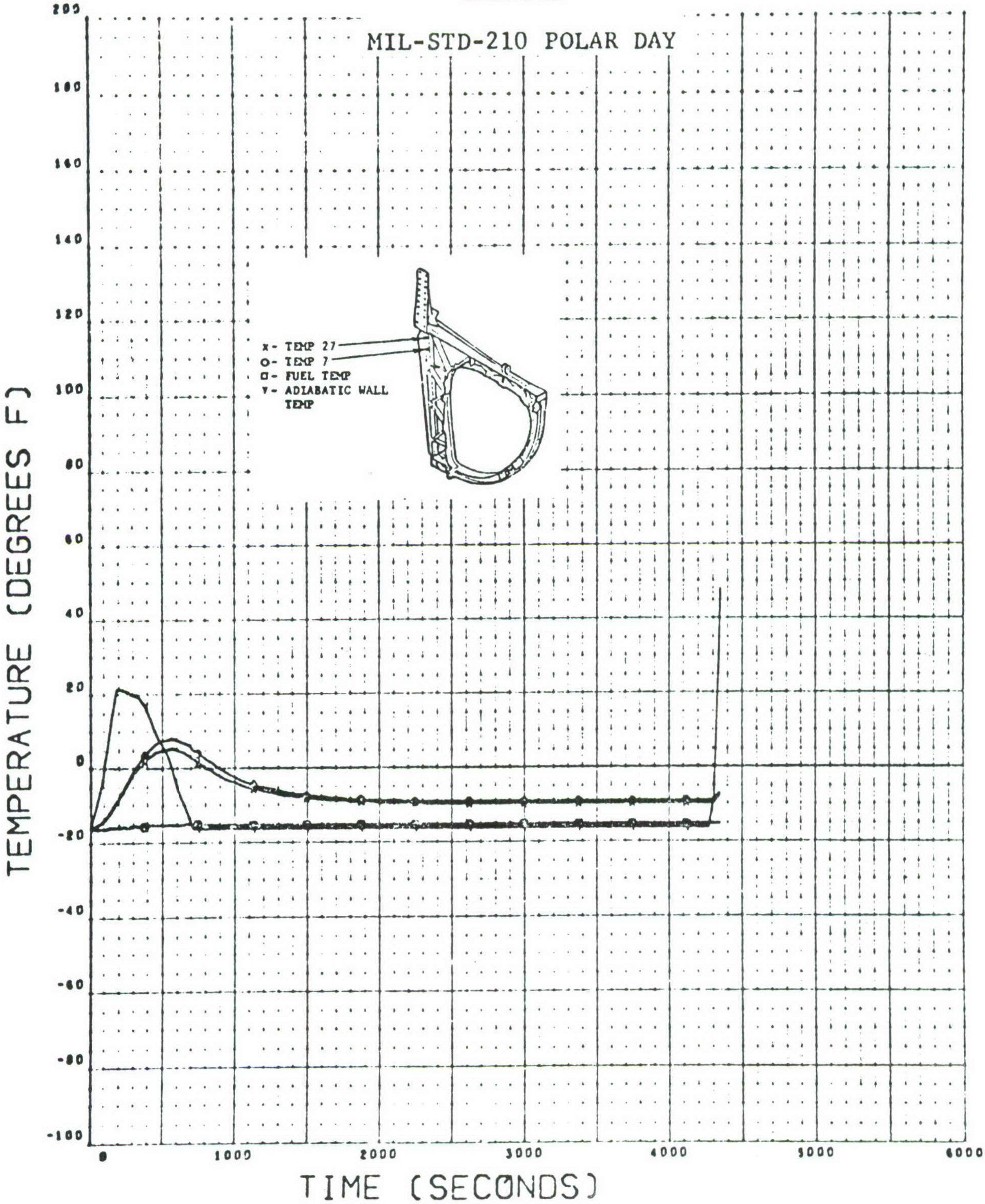


Figure 40

F-111 20^{o} DIVE BOMB RUN
AFTER 1 HOUR MAX LOITER

FSG. STA. 496 BULKHEAD
12B2910

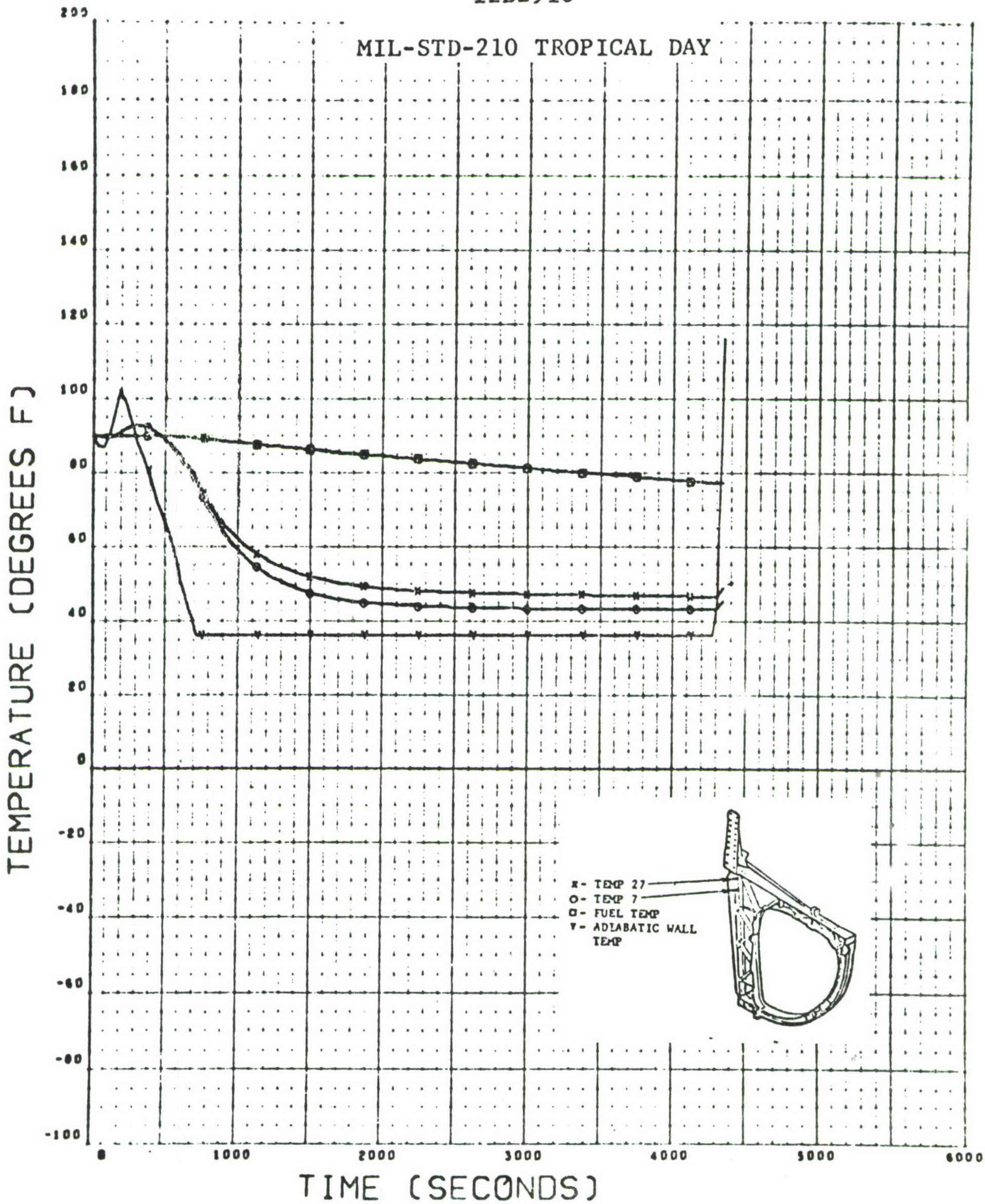


Figure 41

FSG. STA. 496 BULKHEAD
12B2910

F-111 FERRY MISSION

MIL-STD-210 POLAR DAY

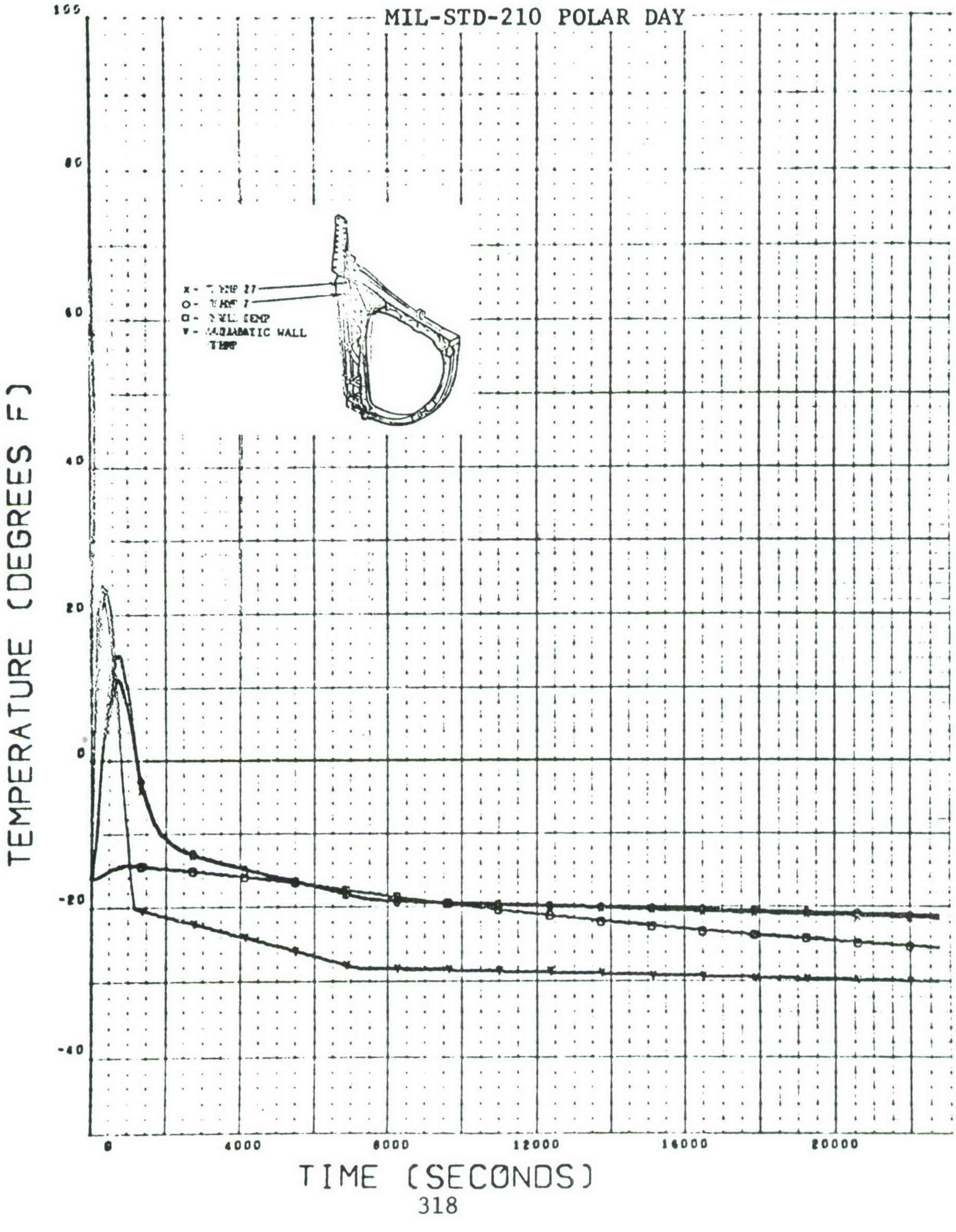


Figure 42

UPPER LONGERON 12B1891
FSG. STA. 523

F-111 TRAINING MISSION TR(A)-2

(M = 1.5 DELETED)

ARDC 1959 STANDARD DAY

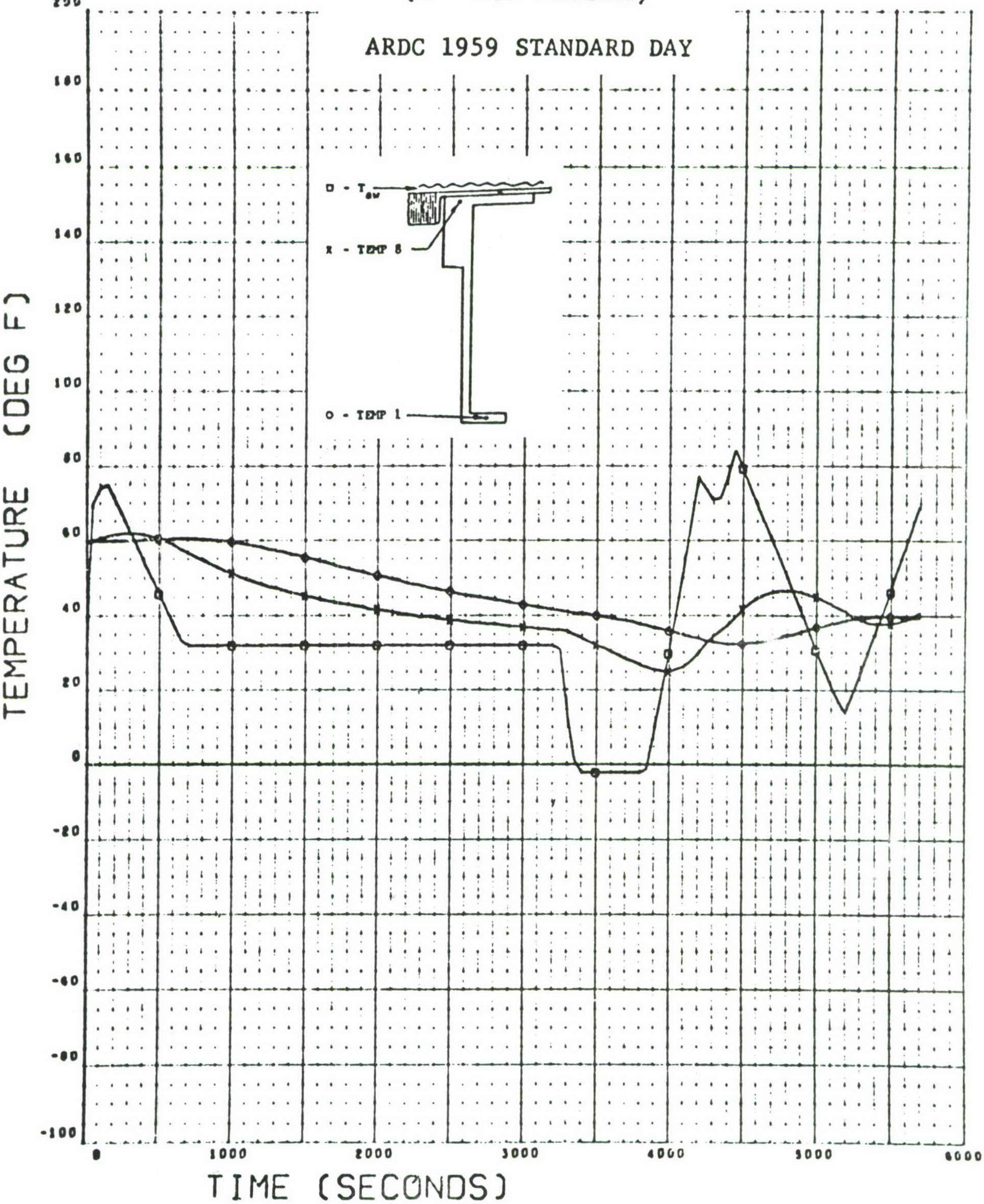


Figure 43

UPPER LONGERON 12B1891
FSG. STA. 523

F-111 TRAINING MISSION TR(A)-2

(M = 1.5 DELETED)

MIL-STD-210 POLAR DAY

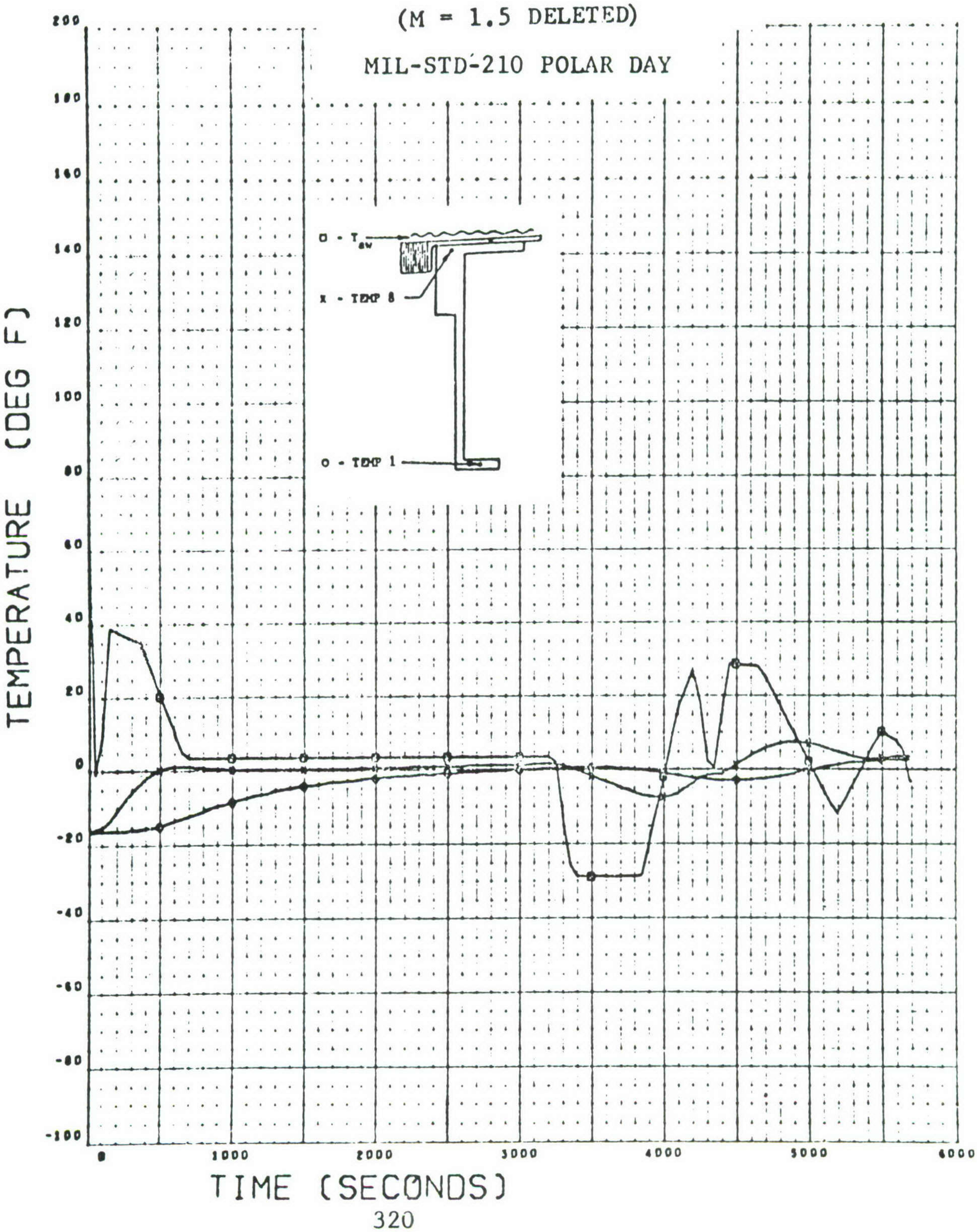


Figure 44

UPPER LONGERON 12B1891
FSG. STA. 523

F-111 TRAINING MISSION TR(A)-2
(M = 1.5 DELETED)
MIL-STD-210 TROPICAL DAY

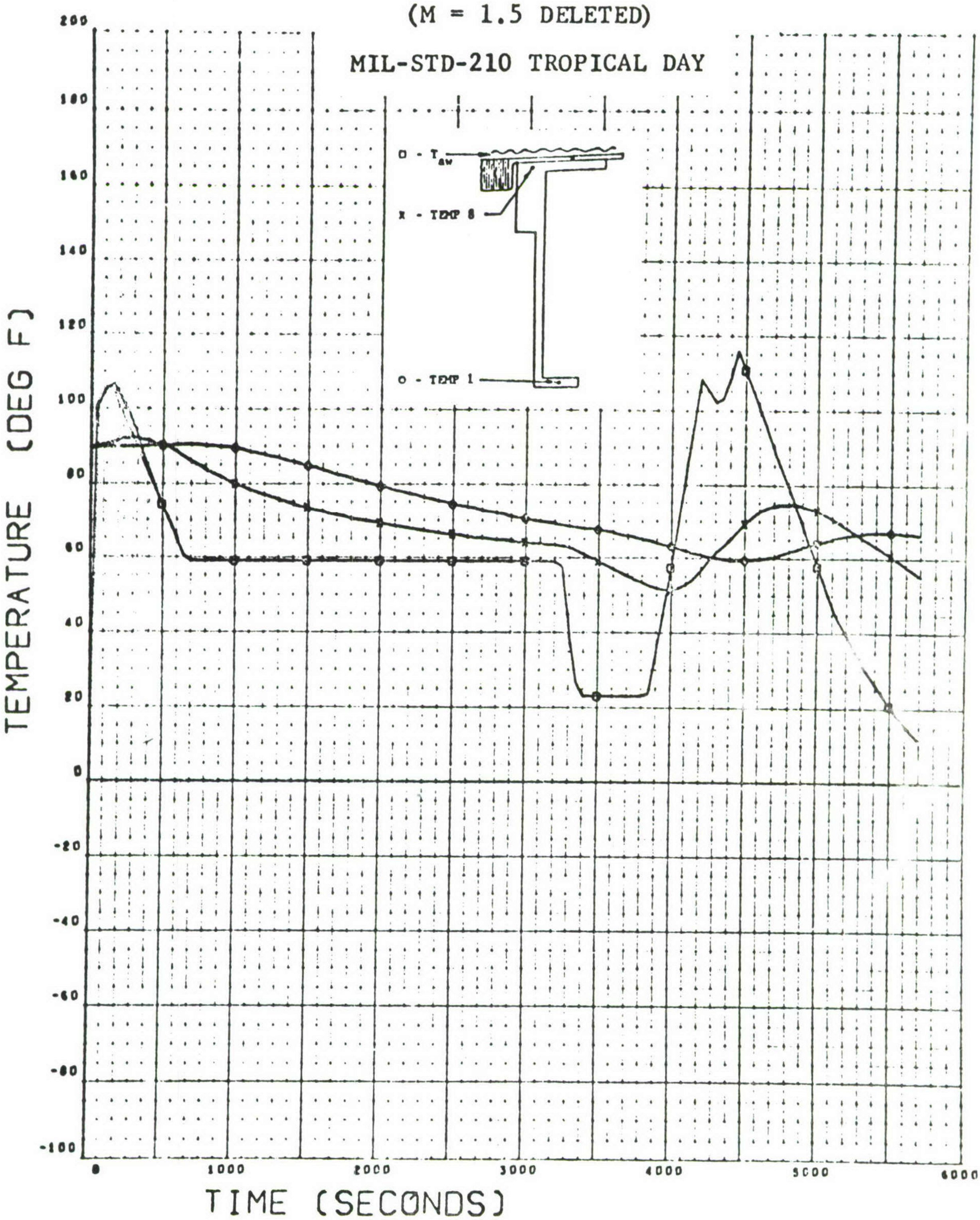


Figure 45

UPPER LONGERON 12B1891
FSG. STA. 523

F-111 TRAINING MISSION TR(A)-5

ARDC 1959 STANDARD DAY

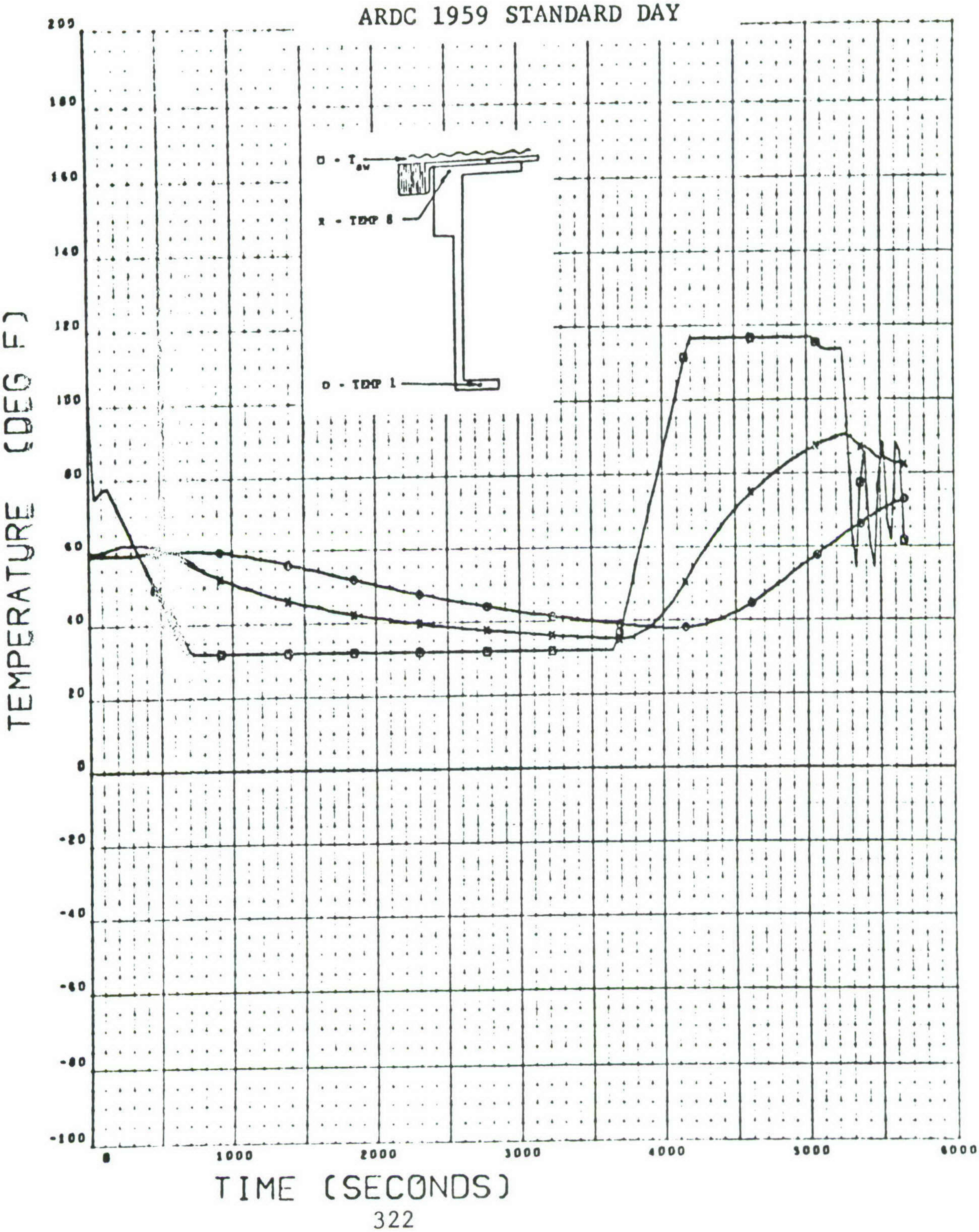


Figure 46

UPPER LONGERON 12B1891
FSG. STA. 523

F-111 TRAINING MISSION TR(A)-5

MIL-STD-210 POLAR DAY

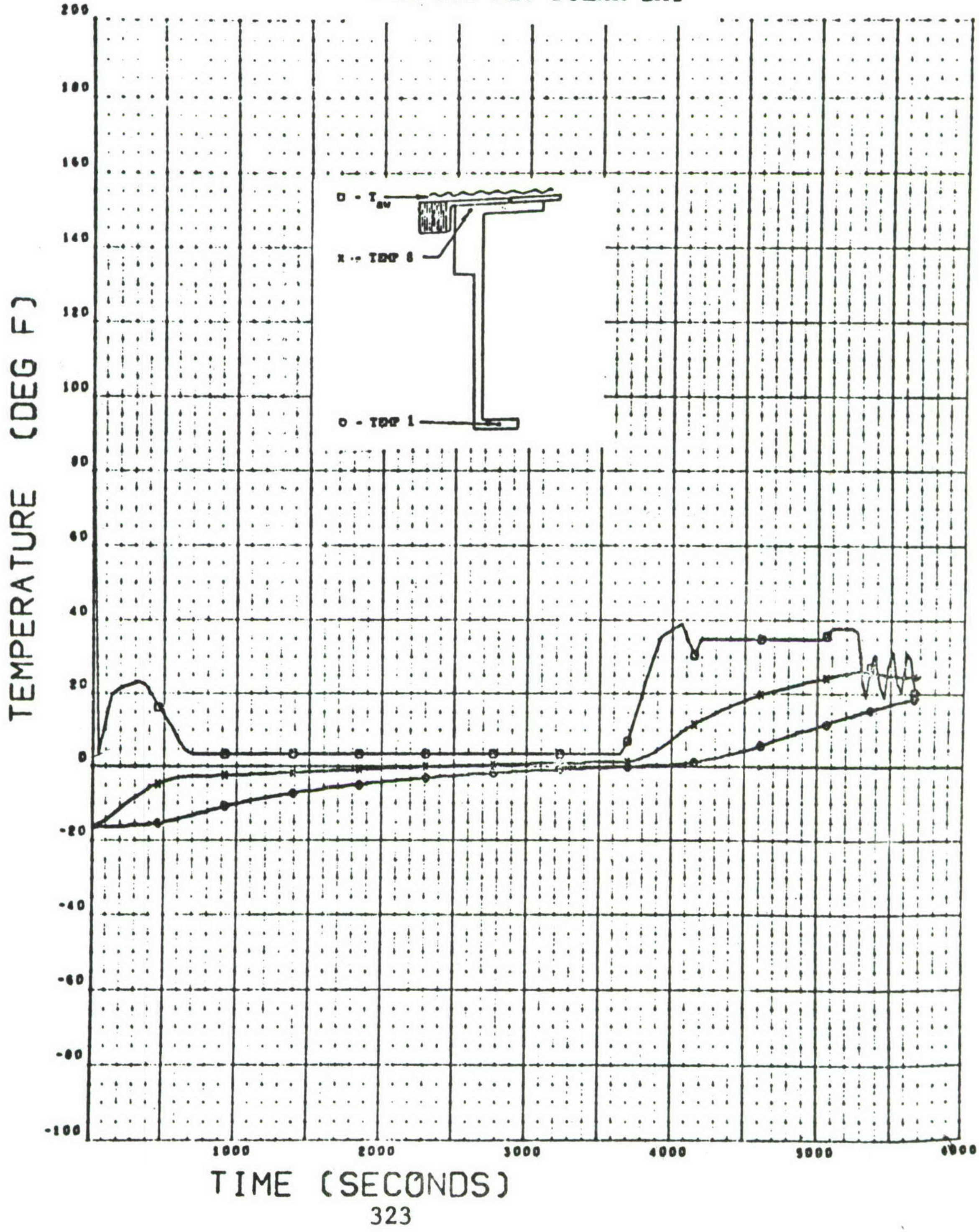


Figure 47

UPPER LONGERON 12B1891
FSG. STA. 523

F-111 TRAINING MISSION TR(A)-5

MIL-STD-210 TROPICAL DAY

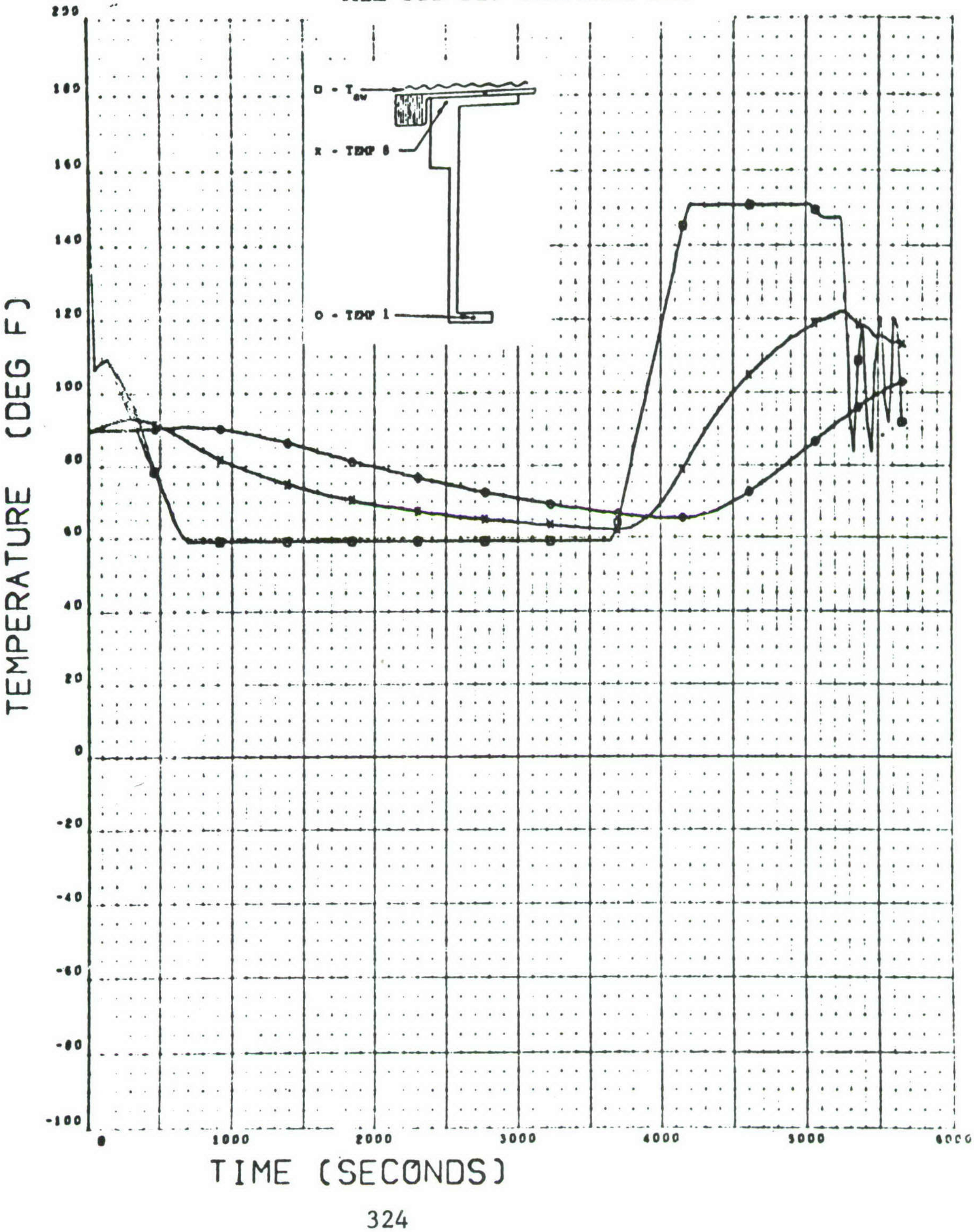


Figure 48

UPPER LONGERON 12B1891
FSG. STA. 523

F-111 20° DIVE BOMB RUN
AFTER 1 HOUR MAX LOITER

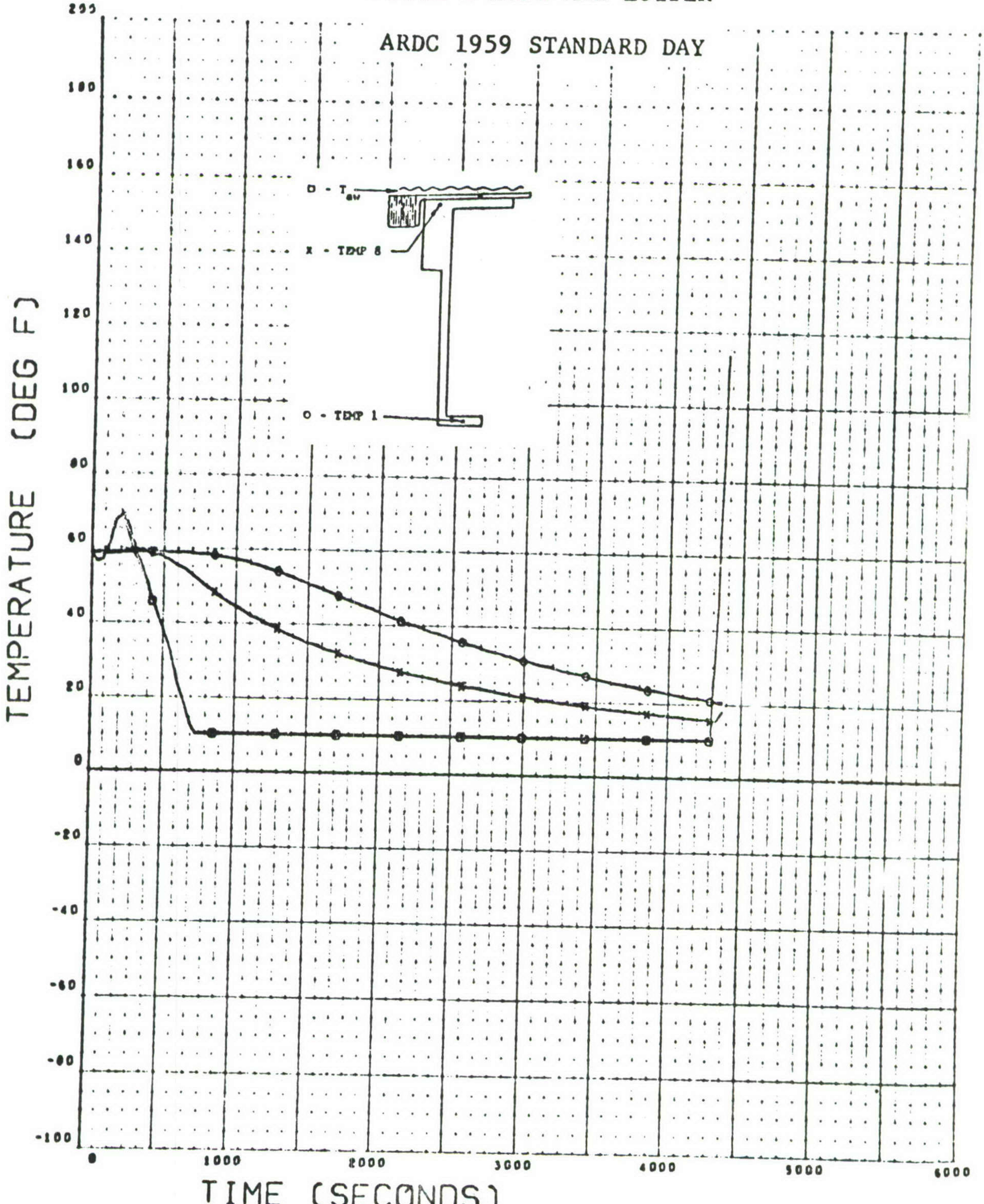


Figure 49

UPPER LONGERON 12B1891
FSG. STA. 523

F-111 20° DIVE BOMB RUN
AFTER 1 HOUR MAX LOITER

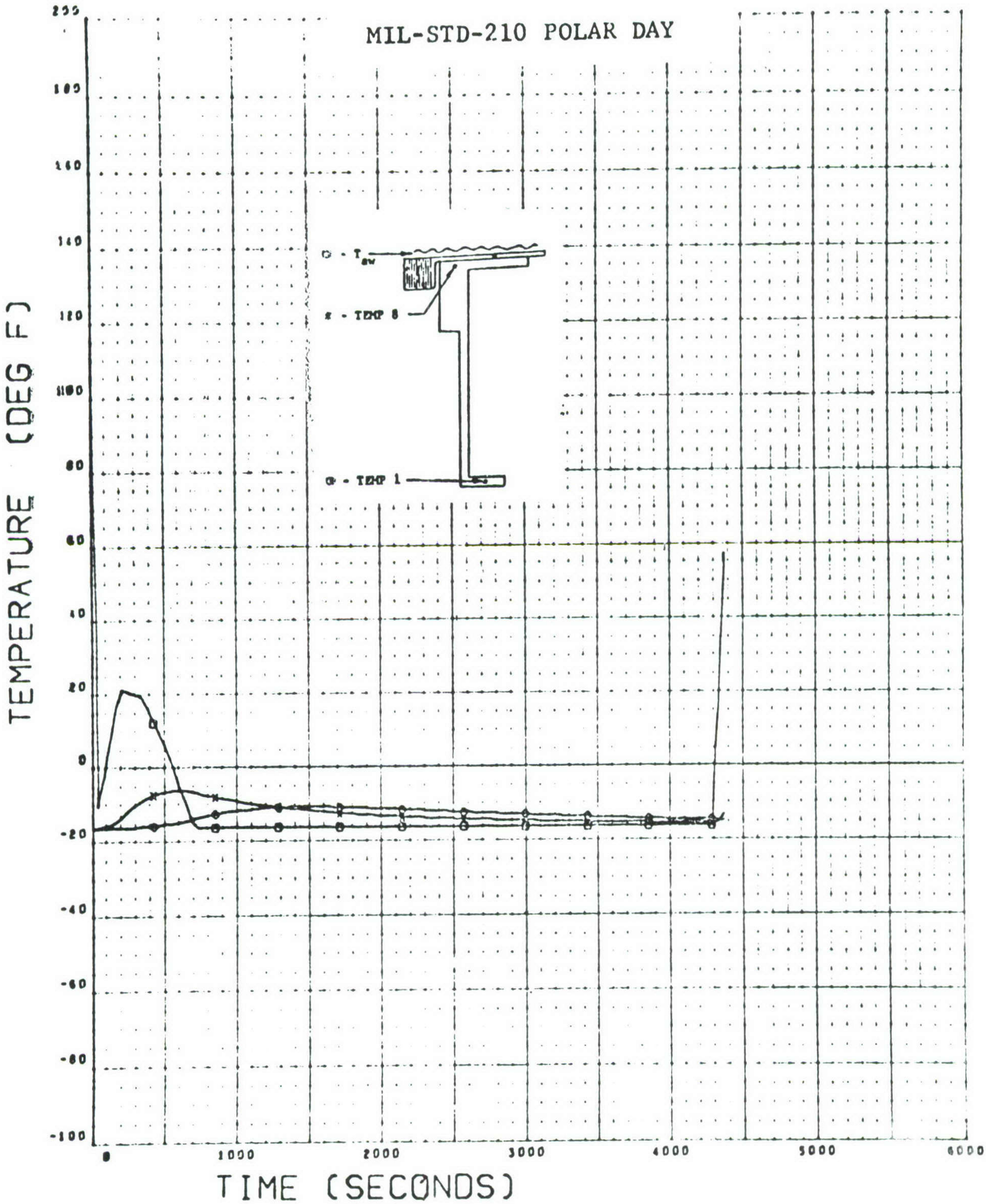


Figure 50

UPPER LONGERON 12B1891
FSG. STA. 523

F-111 20° DIVE BOMB RUN
AFTER 1 HOUR MAX LOITER

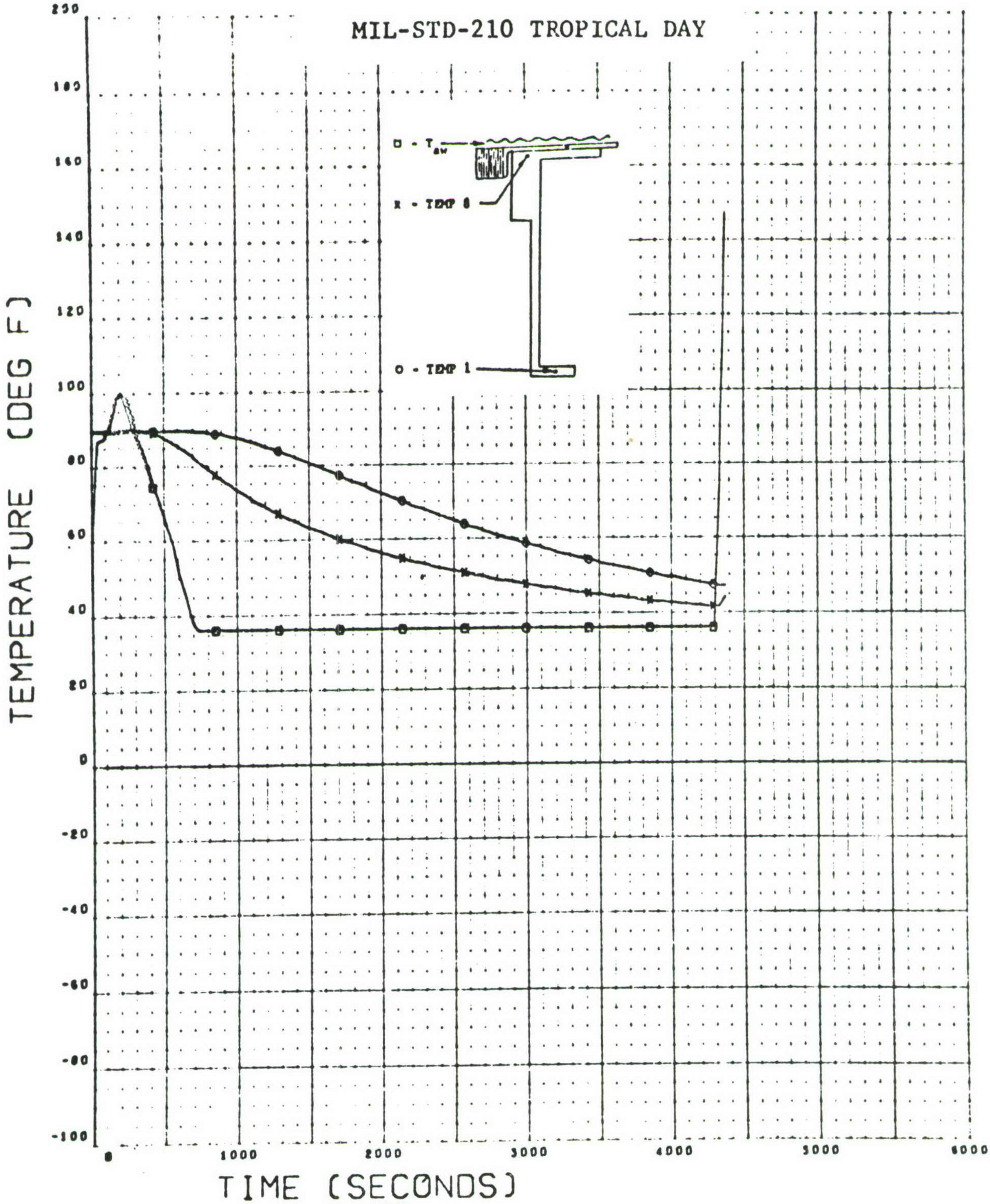


Figure 51

UPPER LONGERON 12B1891
FSG. STA. 523

F-111 FERRY MISSION

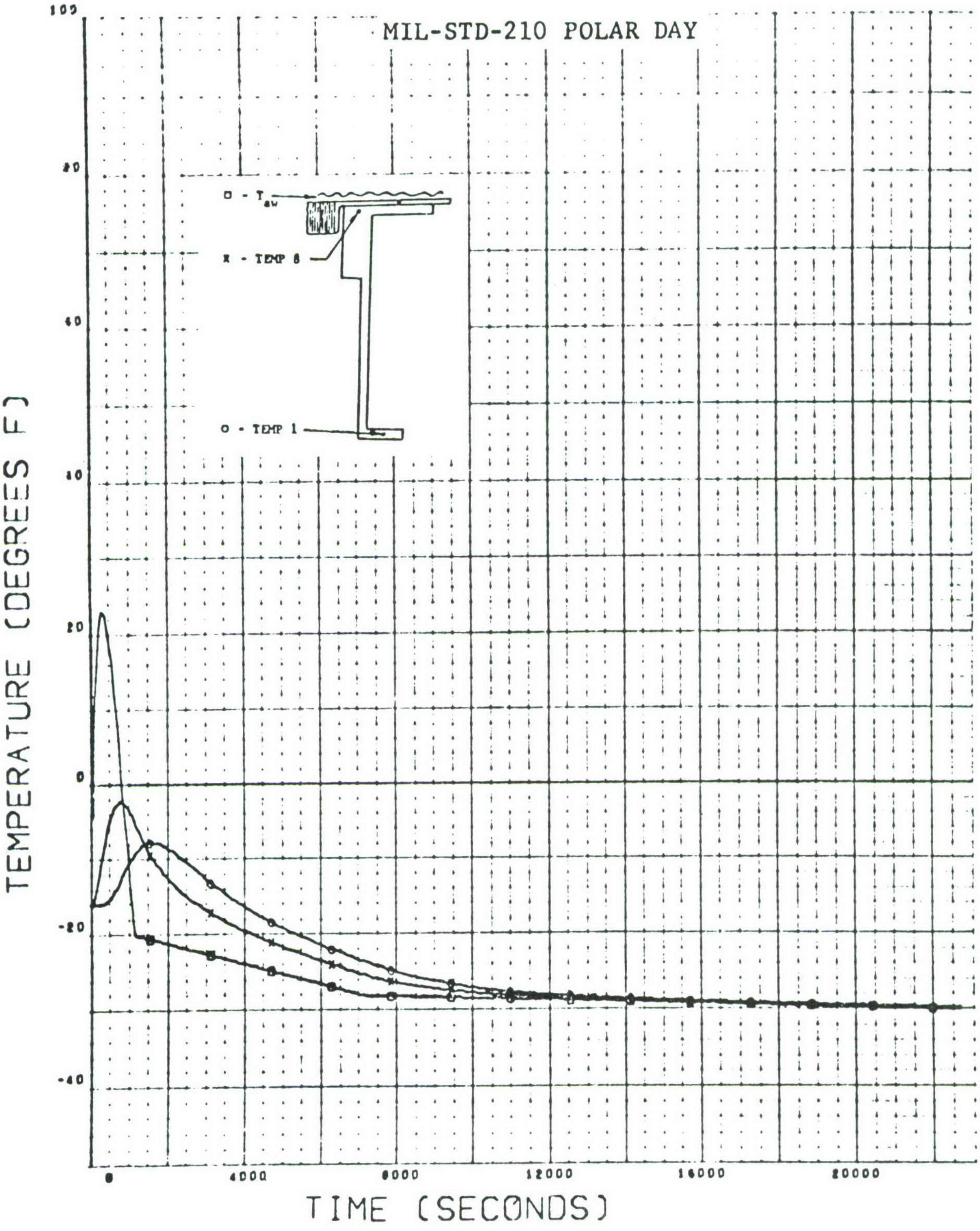


Figure 52

HORIZONTAL TAIL PIVOT SHAFT
FSG. STA. 770 BHD.
12B10521

F-111 TRAINING MISSION TR(A)-2
(M = 1.5 DELETED)

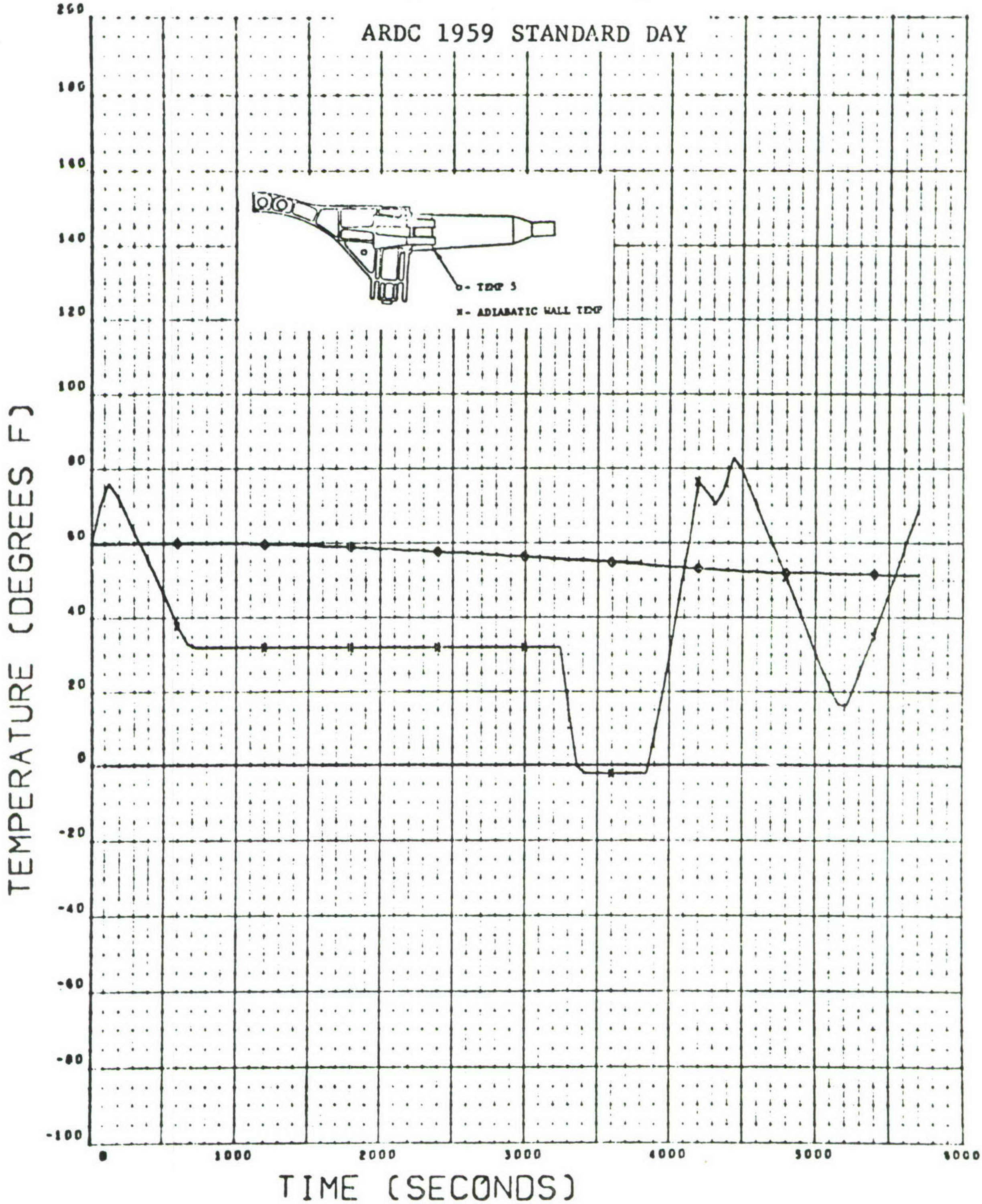


Figure 53

HORIZONTAL TAIL PIVOT SHAFT
FSG. STA. 770 BHD.
12B10521

F-111 TRAINING MISSION TR(A)-2
(M = 1.5 DELETED)

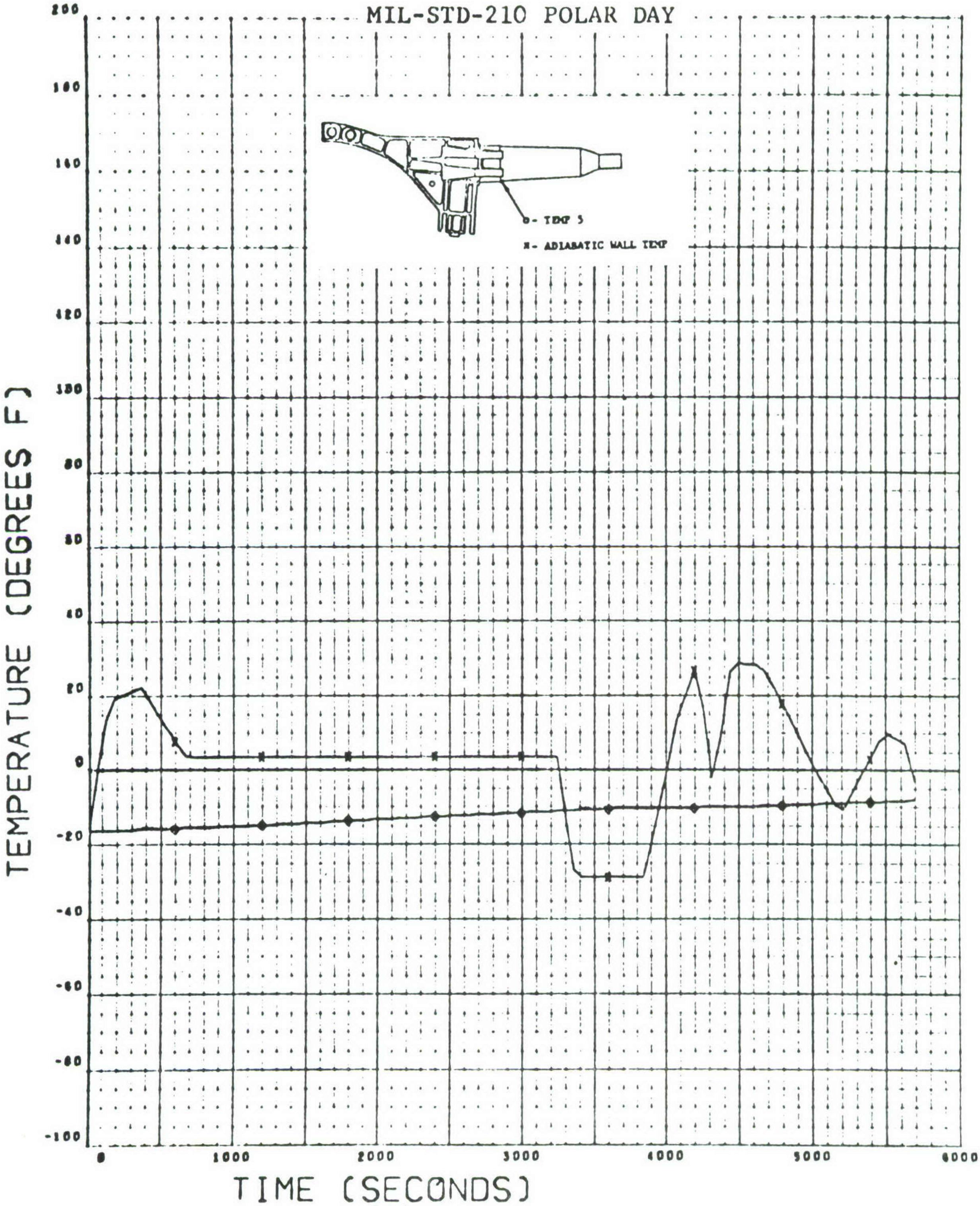


Figure 54

HORIZONTAL TAIL PIVOT SHAFT
FSG. STA. 770 BHD.
12B10521

F-111 TRAINING MISSION TR(A)-2
(M = 1.5 DELETED)

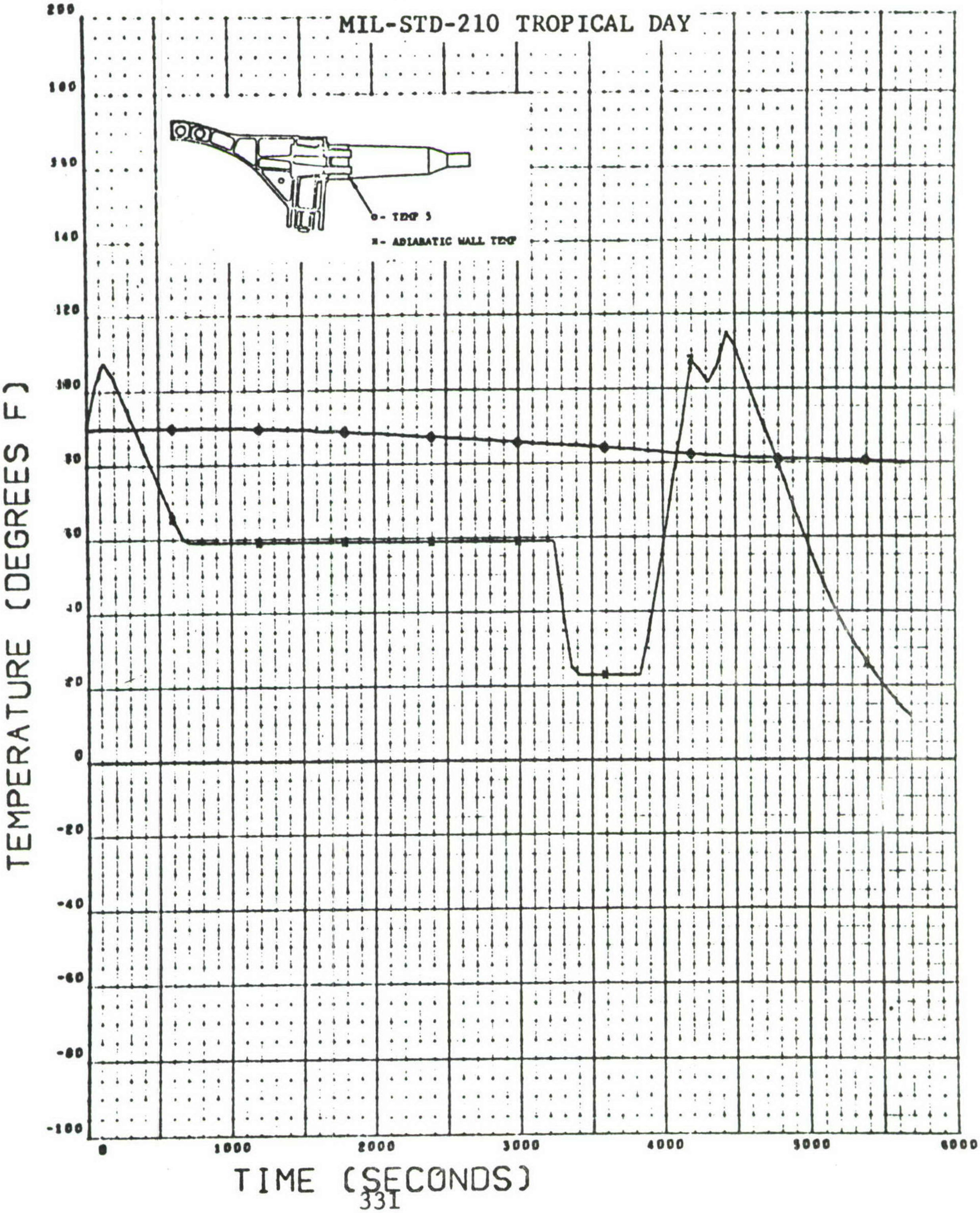


Figure 55

HORIZONTAL TAIL PIVOT SHAFT
FSG. STA. 770 BHD.
12B10521

F-111 TRAINING MISSION TR(A)-5
ARDC 1959 STANDARD DAY

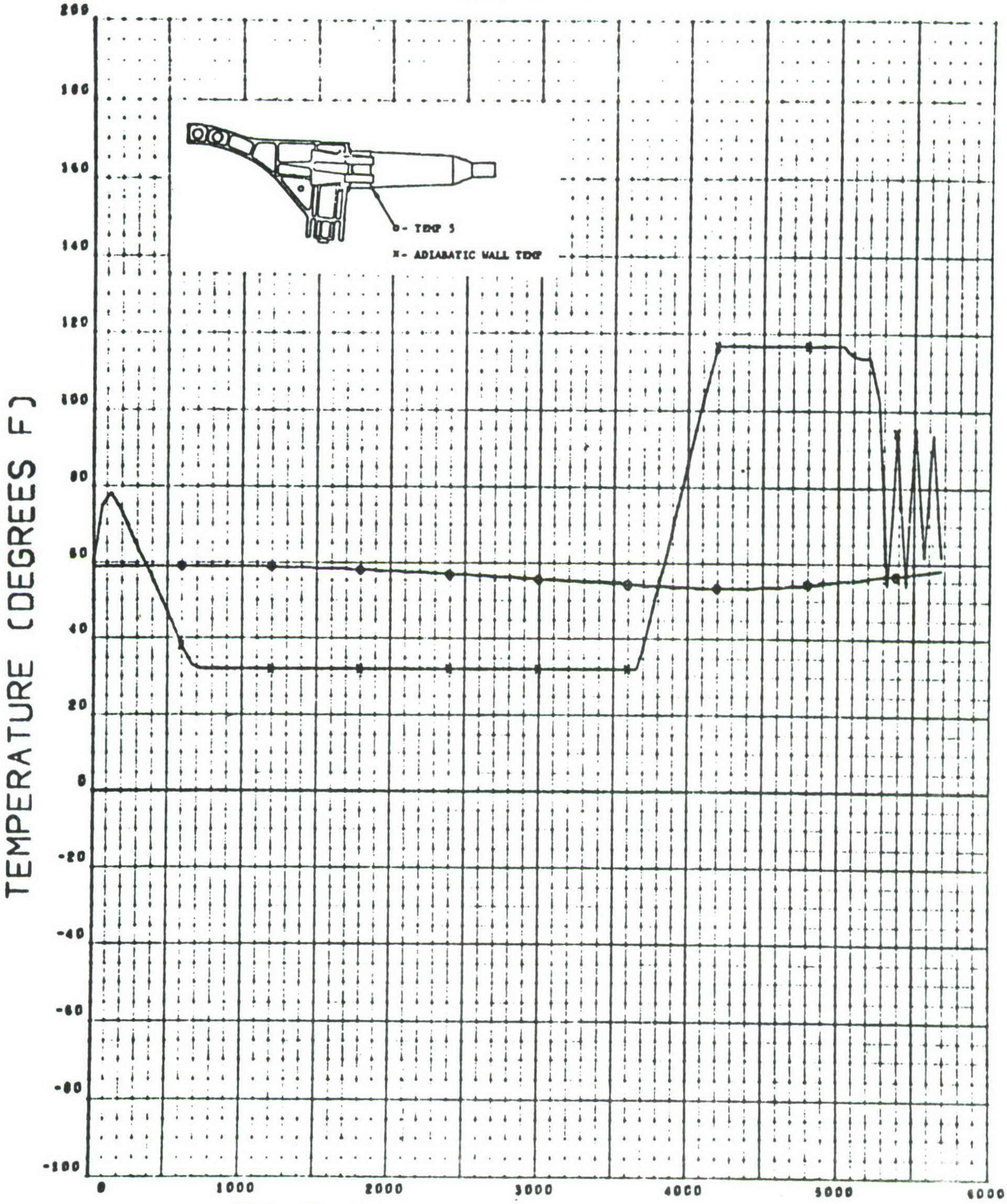


Figure 56

HORIZONTAL TAIL PIVOT SHAFT
FSG. STA. 770 BHD.
12B10521

F-111 TRAINING MISSION TR(A)-5

MIL-STD-120 POLAR DAY

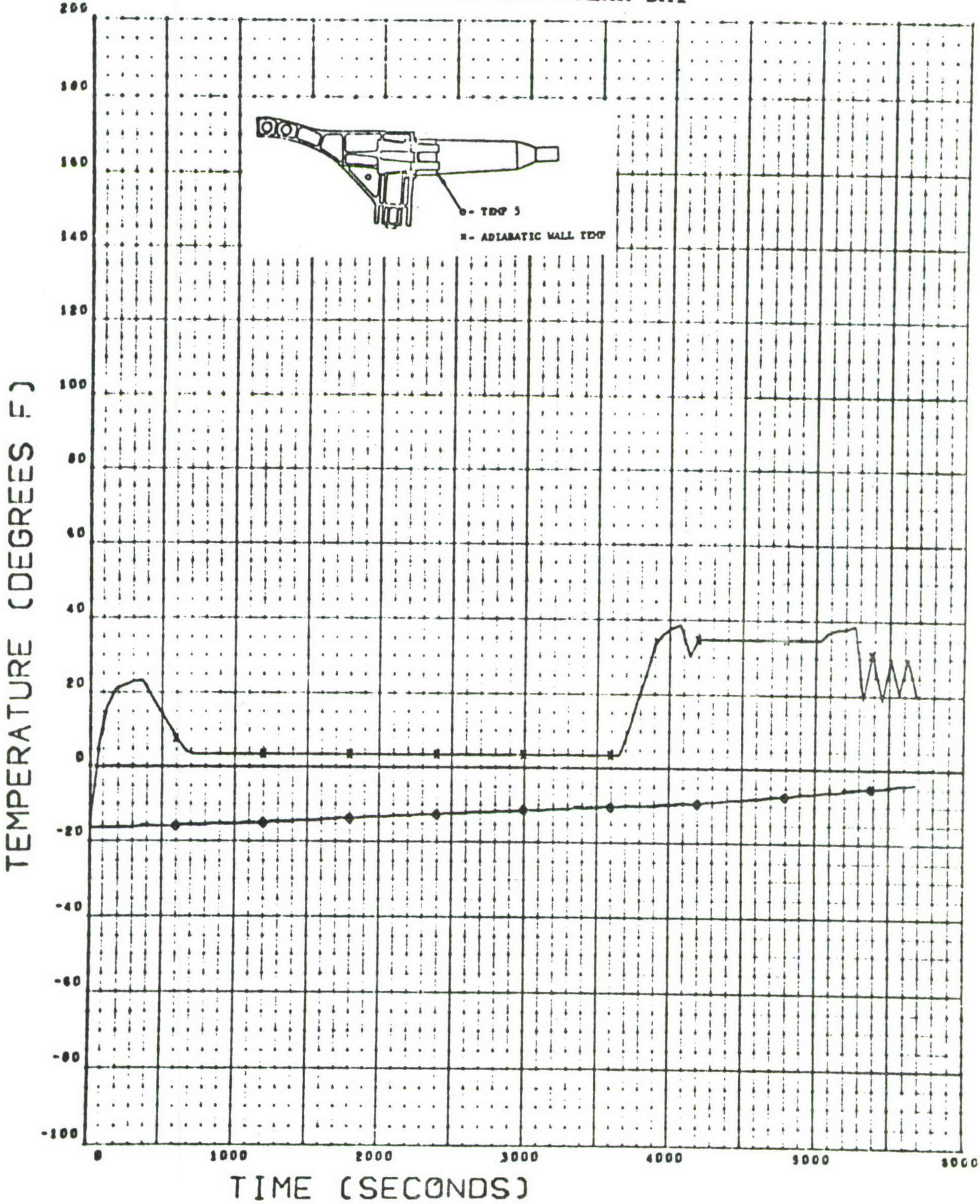


Figure 57

HORIZONTAL TAIL PIVOT SHAFT
FSG. STA. 770 BHD.
12B10521

F-111 TRAINING MISSION TR(A)-5

MIL-STD-210 TROPICAL DAY

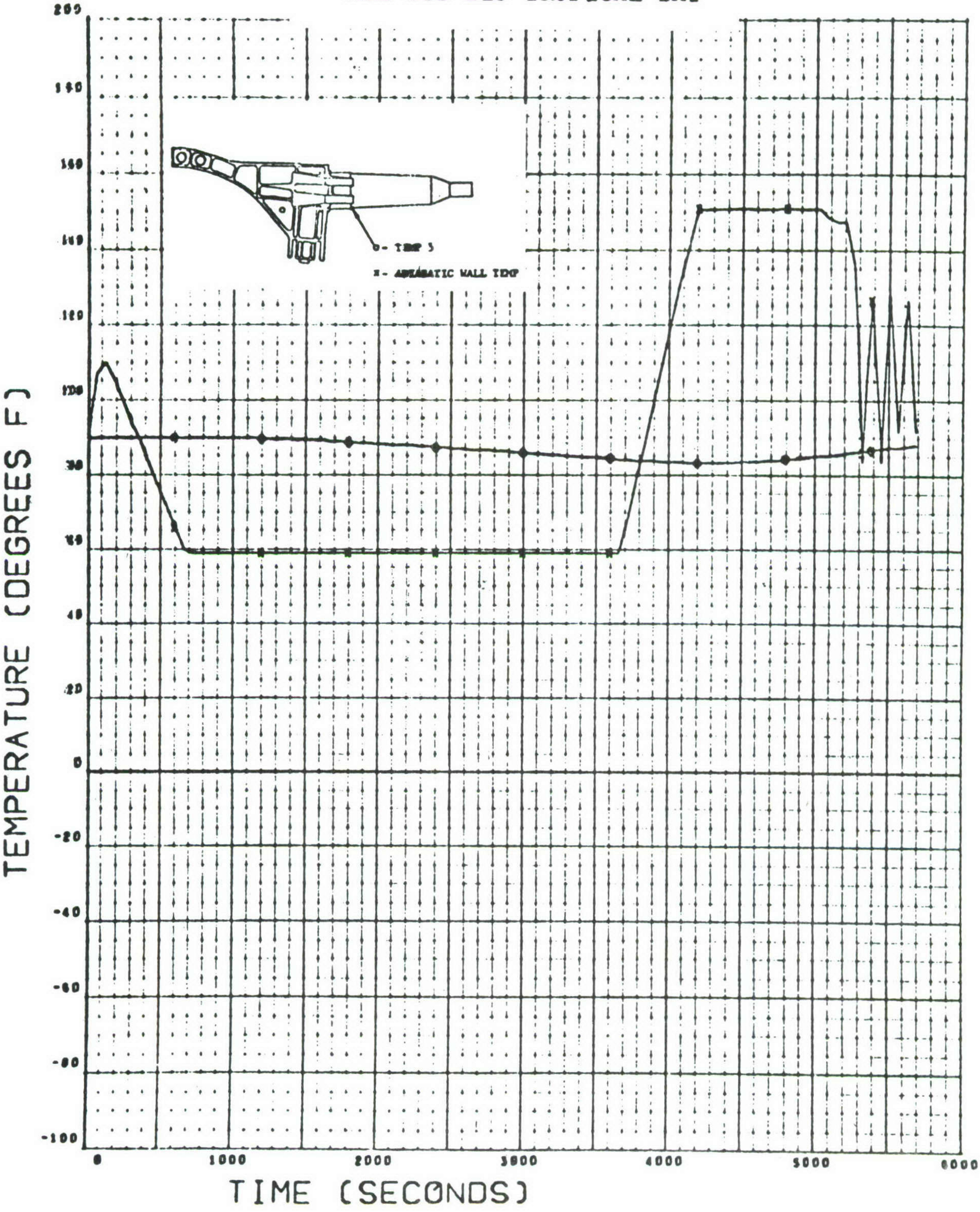


Figure 58

HORIZONTAL TAIL PIVOT SHAFT
FSG. STA. 770 BHD.
12B10521

F-111 20° DIVE BOMB RUN
AFTER 1 HOUR MAX LOITER

ARDC 1959 STANDARD DAY

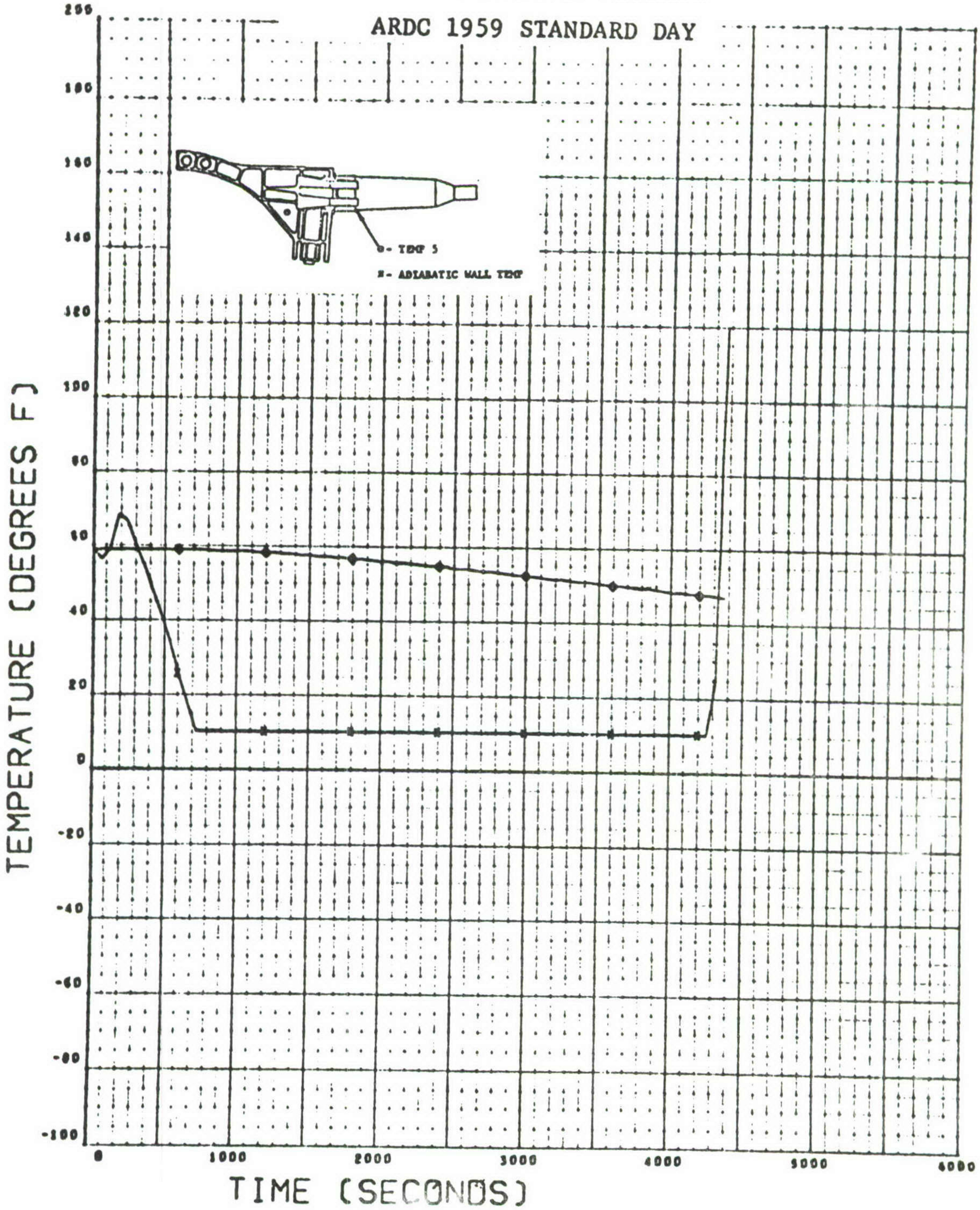


Figure 59

HORIZONTAL TAIL PIVOT SHAFT
FSG. STA. 770 BHD.
12B10521

F-111 20° DIVE BOMB RUN
AFTER 1 HOUR MAX LOITER
MIL-STD-210 POLAR DAY

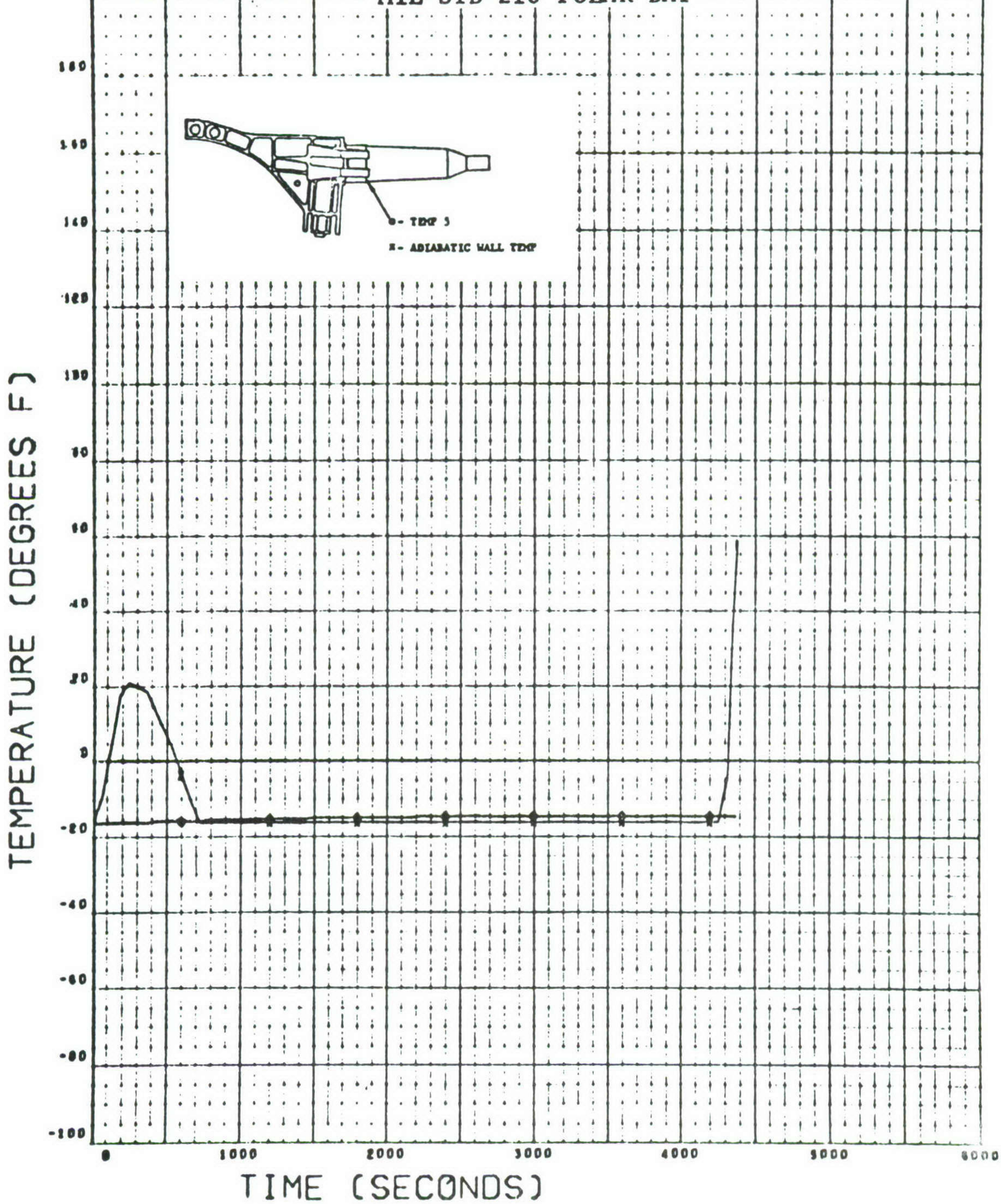


Figure 60

HORIZONTAL TAIL PIVOT SHAFT
FSG. STA. 770 BHD.
12B10521

F-111 20° DIVE BOMB RUN
AFTER 1 HOUR MAX LOITER

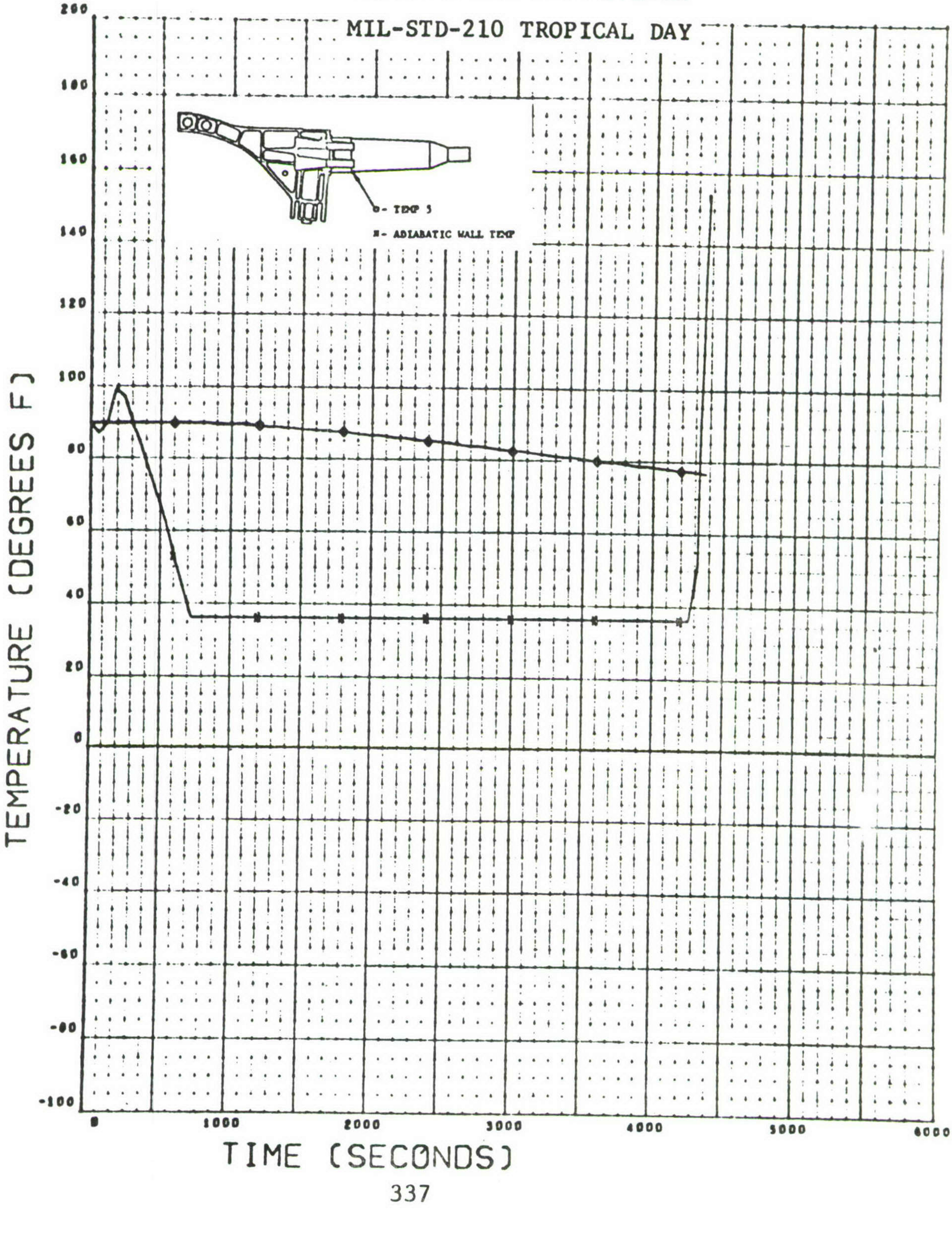


Figure 61

HORIZONTAL TAIL PIVOT SHAFT
FSG. STA. 770 BHD.
12B10521

F-111 FERRY MISSION
MIL-STD-210 POLAR DAY

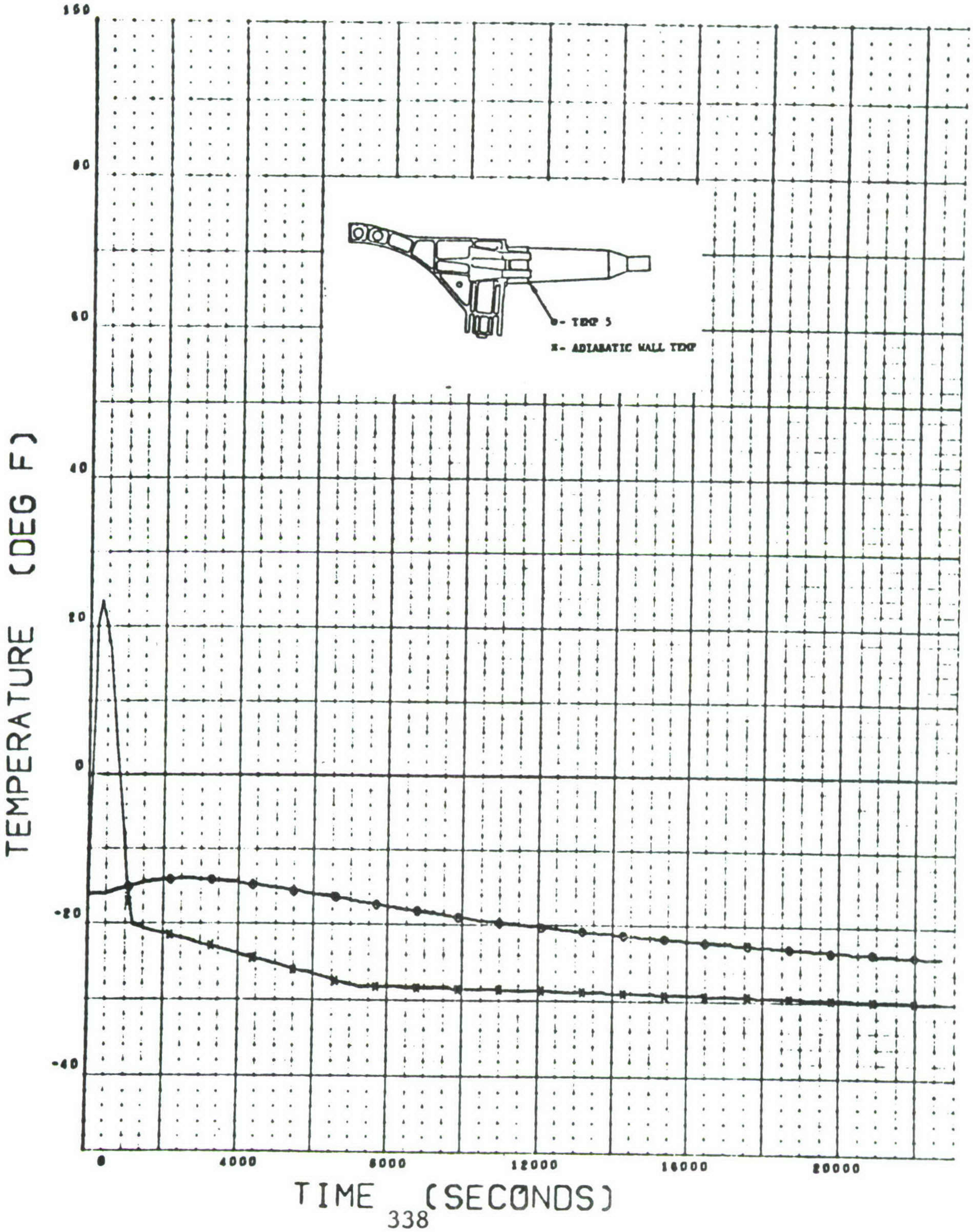


Figure 62

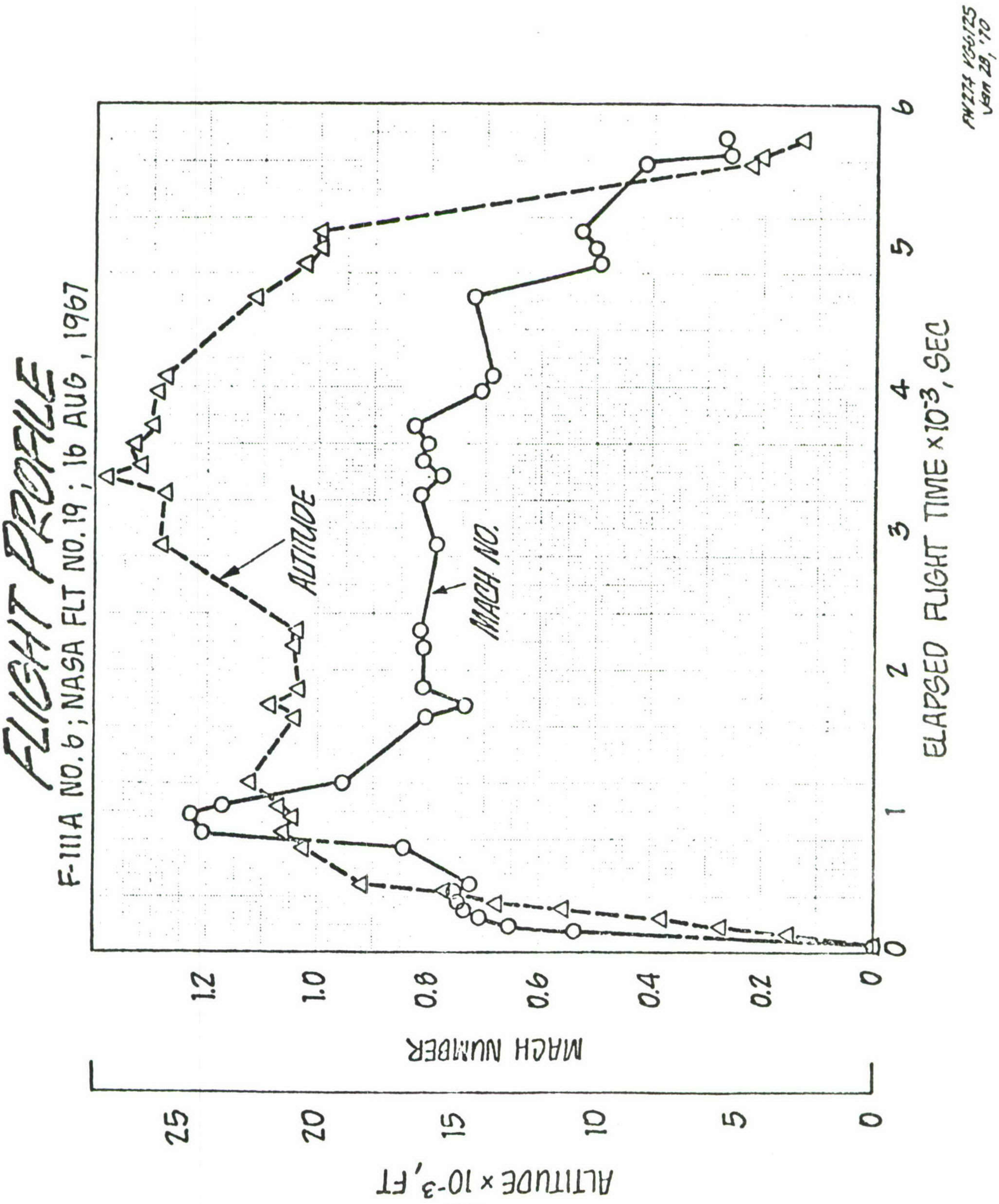
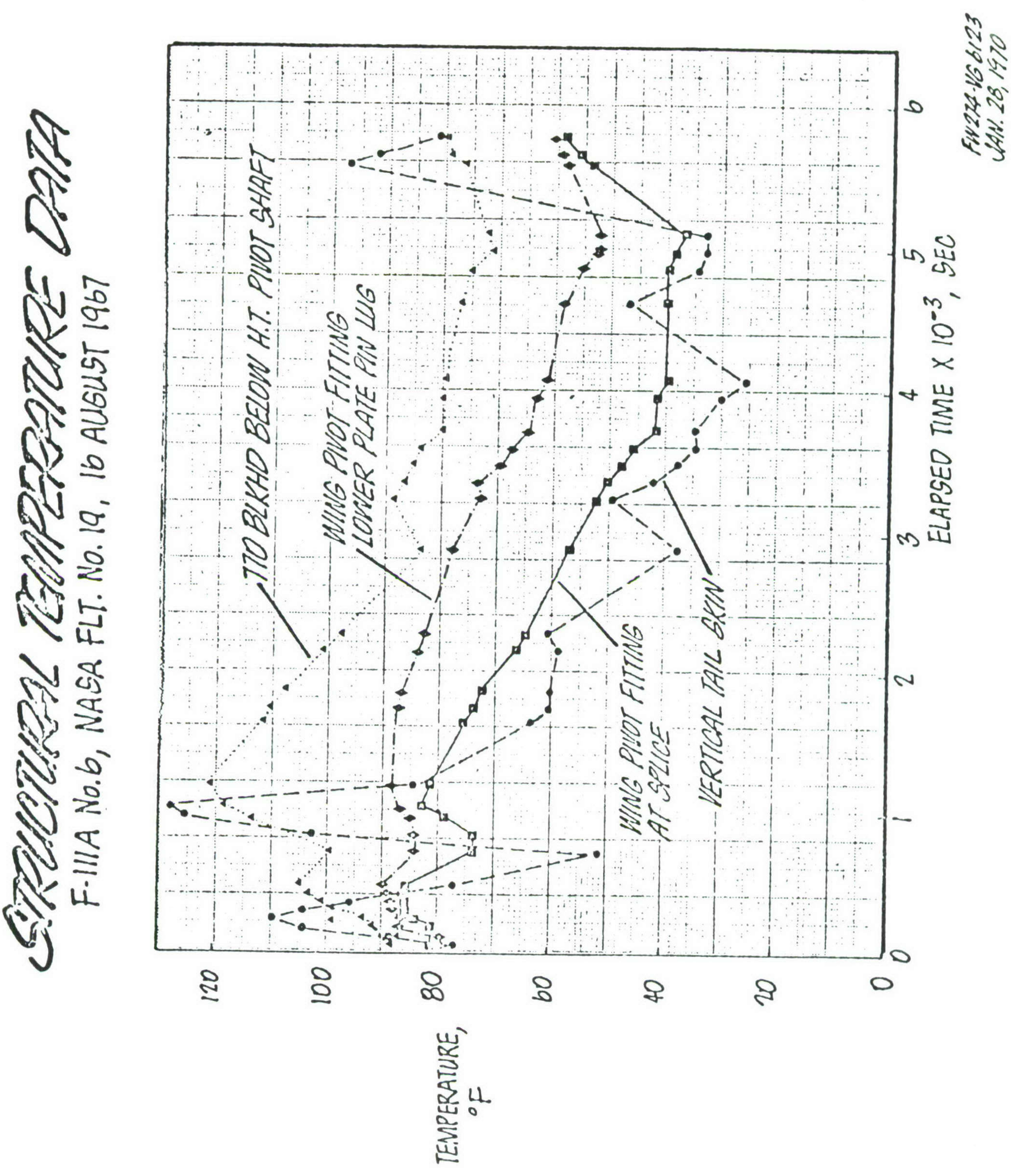

FLIGHT PROFILE
F-111A NO. 6; NASA FLT NO. 19; 16 AUG, 1967
ALTITUDE
MACH NO.
ELAPSED FLIGHT TIME × 10⁻³, SEC
MACH NUMBER
ALTITUDE × 10⁻³, FT
0
1
2
3
4
5
6
0
0.2
0.4
0.6
0.8
1.0
1.2
0
5
10
15
20
25

Figure 63


STRUCTURAL TEMPERATURE DATA
F-111A No.6, NASA FLT. No. 19, 16 AUGUST 1967
770 BLKHD BELOW H.T. PIVOT SHAFT
WING PIVOT FITTING LOWER PLATE PIN LUG
WING PIVOT FITTING AT SPLICE
VERTICAL TAIL SKIN
TEMPERATURE, °F
0
20
40
60
80
100
120
ELAPSED TIME X 10^-3, SEC
0
1
2
3
4
5
6
FW274-VG 6123
JAN. 28, 1970

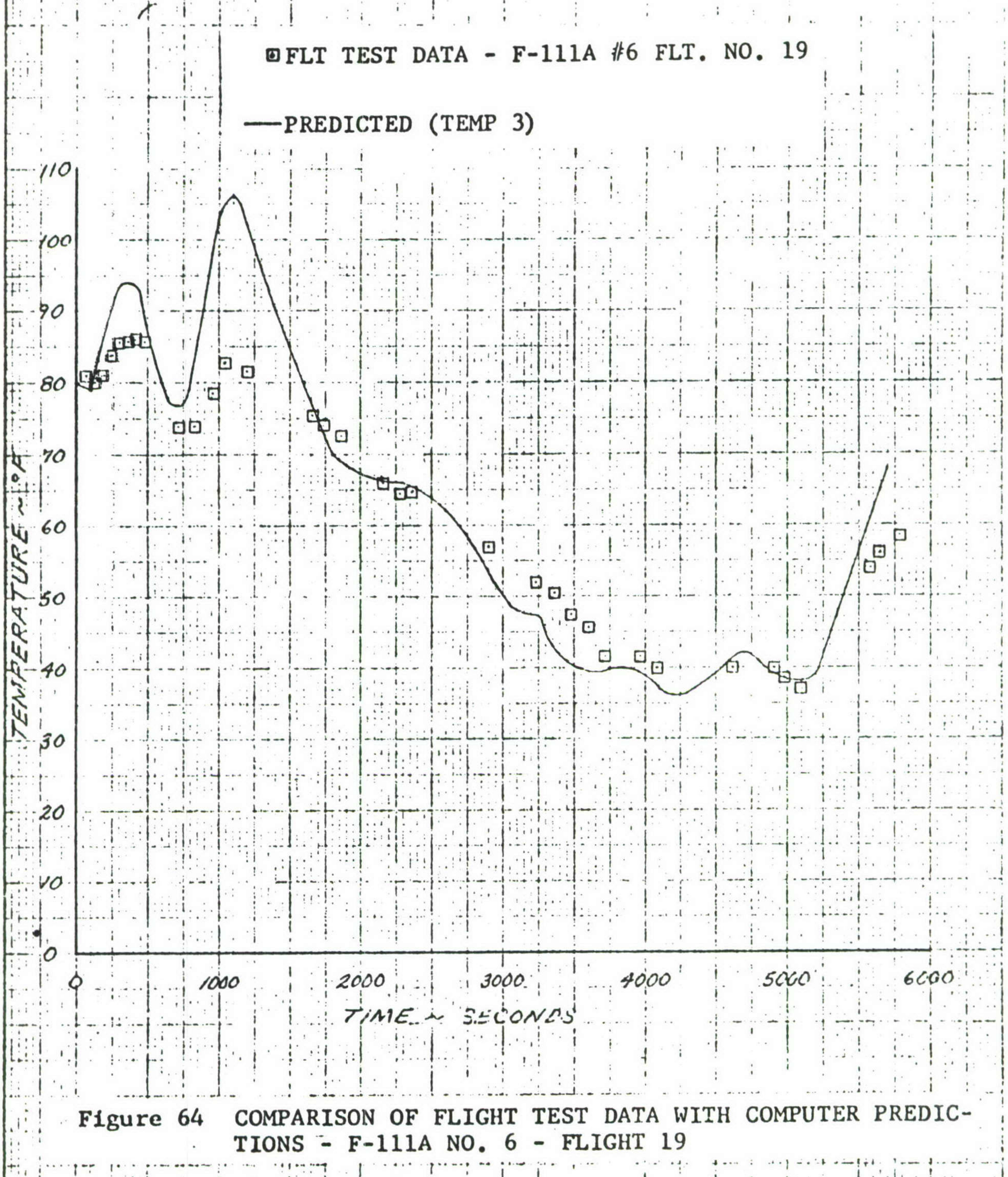


Figure 64 COMPARISON OF FLIGHT TEST DATA WITH COMPUTER PREDICTIONS - F-111A NO. 6 - FLIGHT 19

SUPPLEMENT (C)

MEA-301, A SUMMARY OF NONDESTRUCTIVE INSPECTION PERFORMED ON THE F-111F WING BOX

MEA-301
25 January 1973

A SUMMARY OF NONDESTRUCTIVE INSPECTIONS PERFORMED ON THE F-111F WING BOX STRUCTURE

GENERAL DYNAMICS
Convair Aerospace Division
P. O. Box 748, Fort Worth, Texas 76101

A SUMMARY OF NONDESTRUCTIVE INSPECTIONS PERFORMED ON F-111F WING BOX STRUCTURE

Authorization

Fracture Mechanics for an Advanced
Air Superiority Fighter Wing Structure
(W.O. 751-58-501)

CONTENTS

LIST OF FIGURES

LIST OF TABLES

1.0 SCOPE

Review the fabrication/manufacturing operations of the F-111F wing box detail parts, assembly, and cold proof test to determine the type, sequencing, frequency, and flaw detection criteria of the NDI's experienced.

2.0 SUMMARY

Each of the detail parts, aside from dimensional inspections, received one penetrant inspection per NDTS 10.00, Liquid Penetrant Inspection. Each part, except 12W974 (titanium) received one hardness test per NDTS 15.00, Hardness Testing, Method of Inspection. The wing spar raw material received an ultrasonic inspection per NDTS 50.00, Ultrasonic Inspection, Method of. At assembly the wing box structure received a radiographic inspection per NDTS 30.00-7, X-ray Inspection of F-111 Wing. No further NDI's are accomplished in system operations or cold proof test, although a visual examination is conducted on the exterior of the wing skins after proof test.

3.0 REVIEW CRITERIA

3.1 REVIEW DESCRIPTION

The review was limited to details and assembly of the wing box proper. The wing pivot assembly, flight control structure, and pylon housings were not included.
The following part numbers and nomenclature identify the considered detail parts:

12W950	Wing skin, upper
12W951	Wing skin, lower
12W908	Front spar
12W902	Fwd aux spar
12W903	Center spar
12W904	Aft aux spar
12W905	Rear spar
12W972	Doubler
12W974	Doubler
12W986	Doubler
12W973	Doubler
12W915	Bulkhead # 5
12W914	Bulkhead # 4
12W926	Bulkhead # 3
12W919	Bulkhead # 3.5
12W918	Bulkhead # 2.5
12W821	Bulkhead, Outer housing
12W820	Bulkhead, Inner housing
12W912	Bulkhead # 2
12W917	Bulkhead # 1.7
12W823	Bulkhead, Pylon housing
12W824	Pylon housing support

12W982	Splice	12W822	Bulkhead
12W985	Web spar	12W920	Bulkhead # 1.5
12W983	Flange	12W911	Bulkhead # 1.0
12W988	Splice	12W916	Bulkhead # 0

3.2 TYPICAL PARTS SELECTED

Coordination with Manufacturing Engineering revealed that within each nomenclature family the manufacturing processes experienced were the same; therefore, one representative part was selected from each nomenclature family.
They were:

12W985-9/-10	Web spar	12W908-25/-26	Front spar
12W950-9/-10	Upper wing skin	12W974-9/-10	Doubler
12W915-15/-16	Bulkhead		

In addition, the assembly of the wing box, Items 62, 61, and 60 was reviewed along with 12AEI-11-1047B, Cold Temperature Proof Load Test of F-111F Aircraft.

4.0 FINDINGS

Figures 1 through 6 illustrate the principle manufacturing steps and inspections performed on the parts from fabrication to delivery. The NDTS used and its sequence is also shown. Table 1 summarizes the NDI experience.

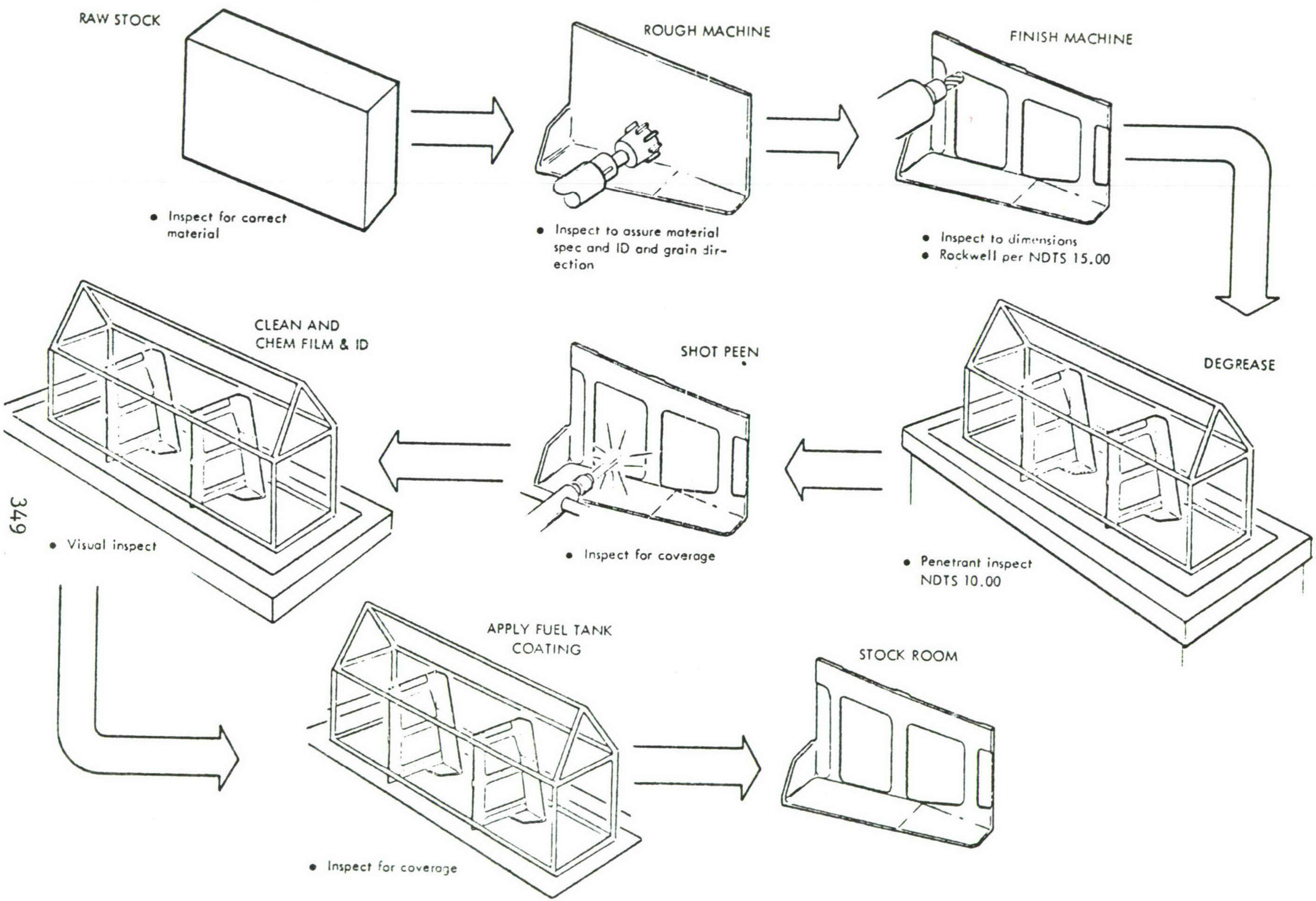


Figure 1 12W985-9/-10 WEB SPAR, 2024-T851 MATERIAL(FMS1010-T851)

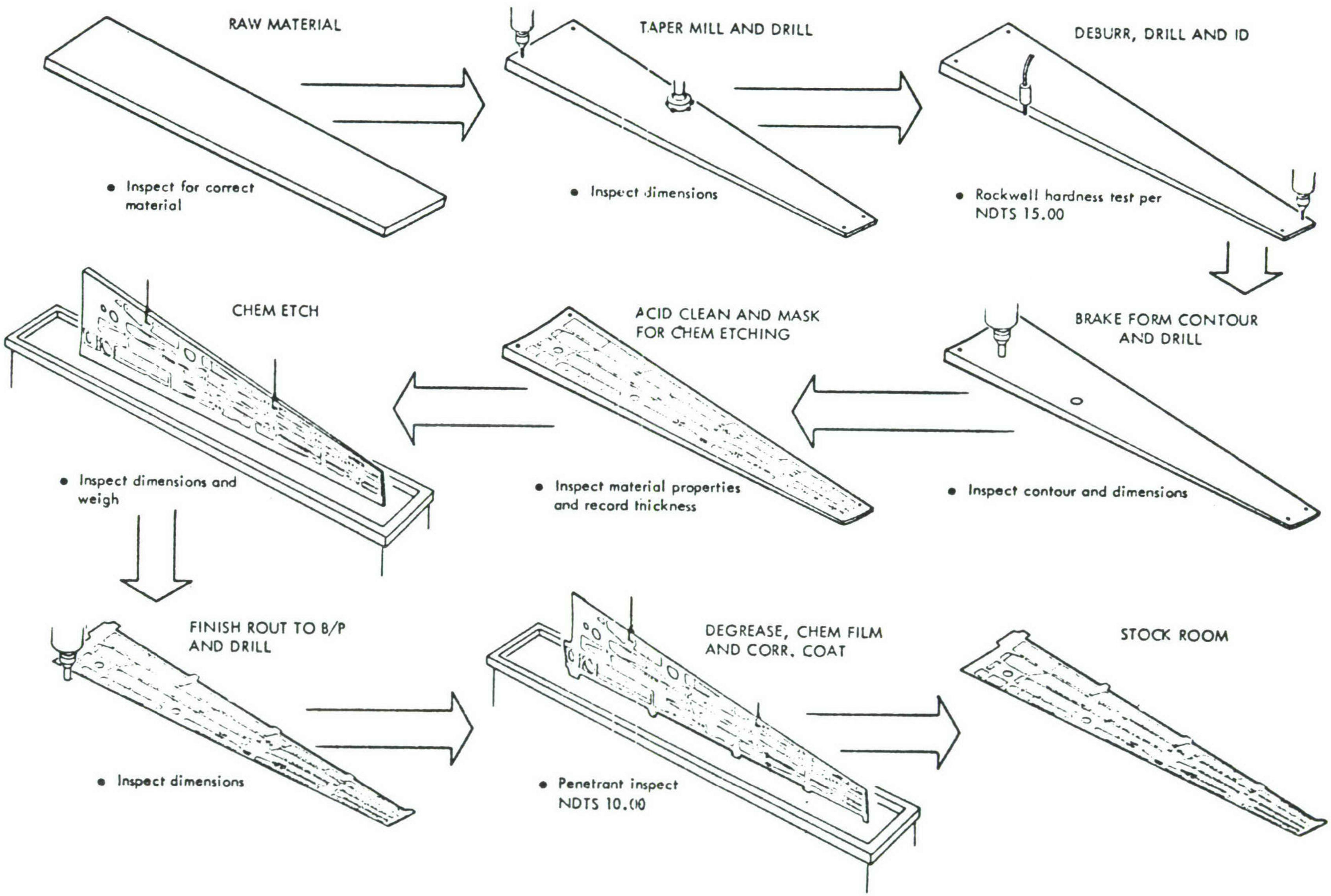


Figure 2 12W950-9/-10 UPPER WING SKIN, 2024 A1 PLATE (QQA 355-T351)

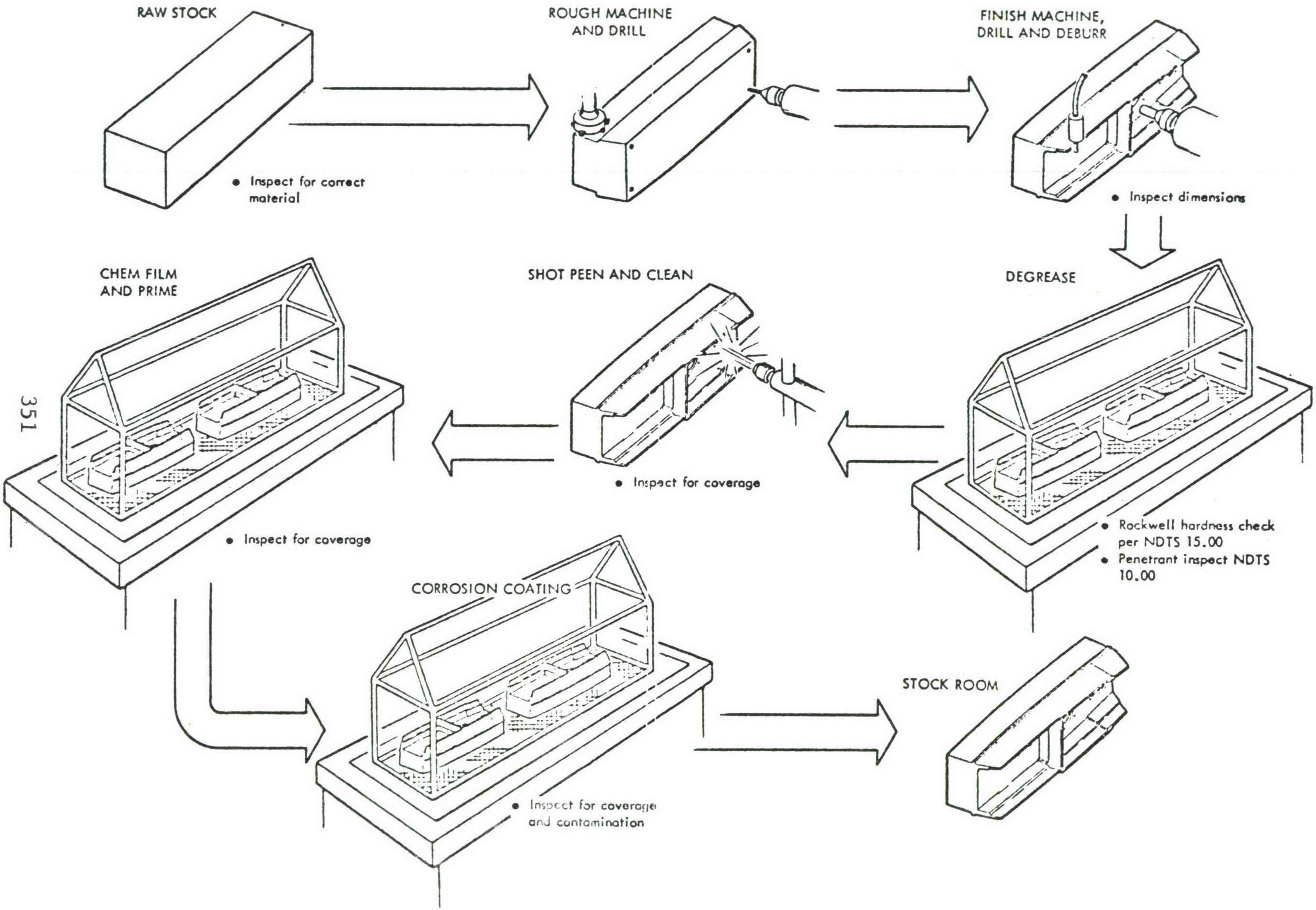


Figure 3 12W915-15/-16 BULKHEAD #5
2024-T851 Al (FMS 1010)

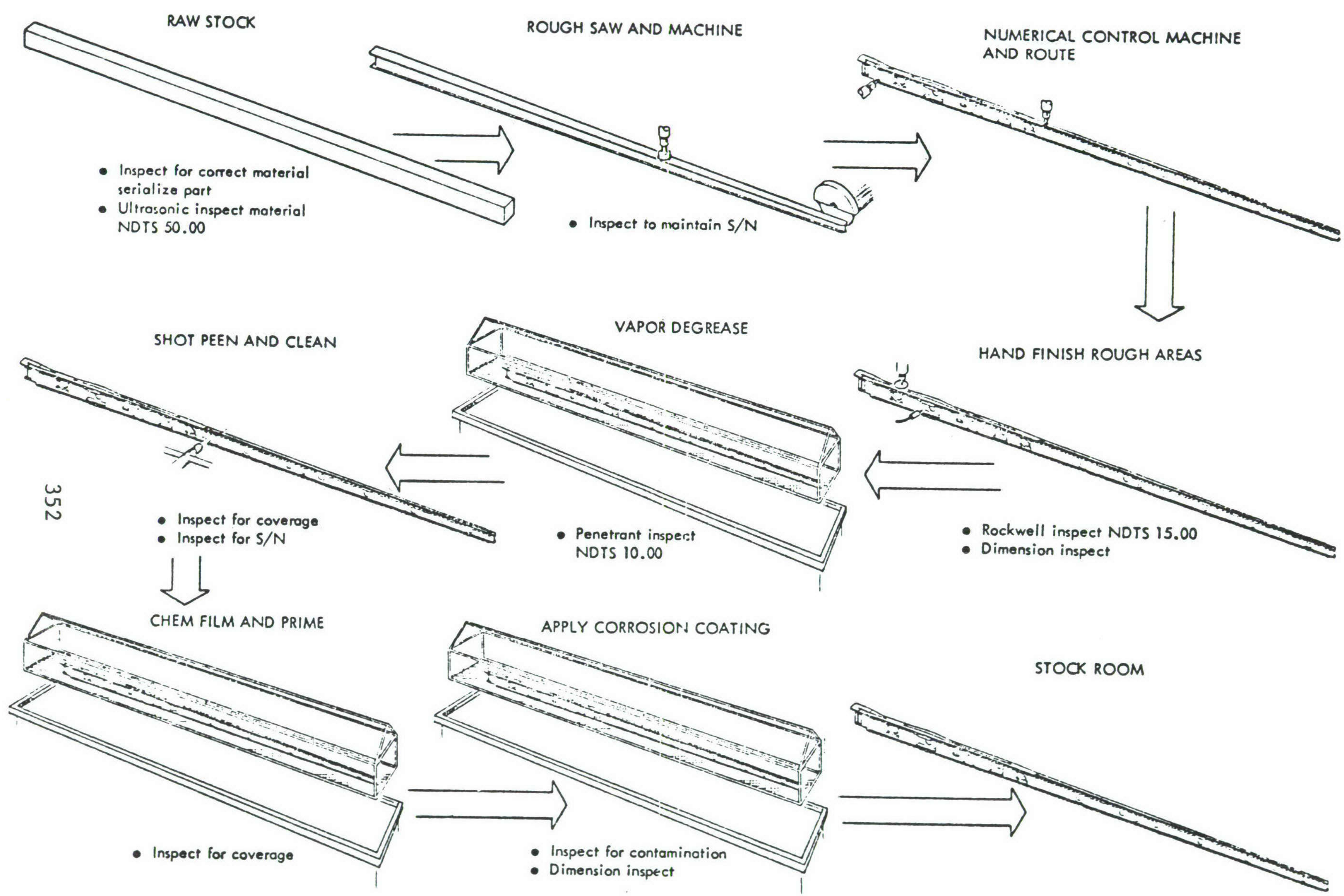


Figure 4 12W908-25/-26 FRONT SPAR, 2024-T851 A1 (FMS 1010 FLAT)

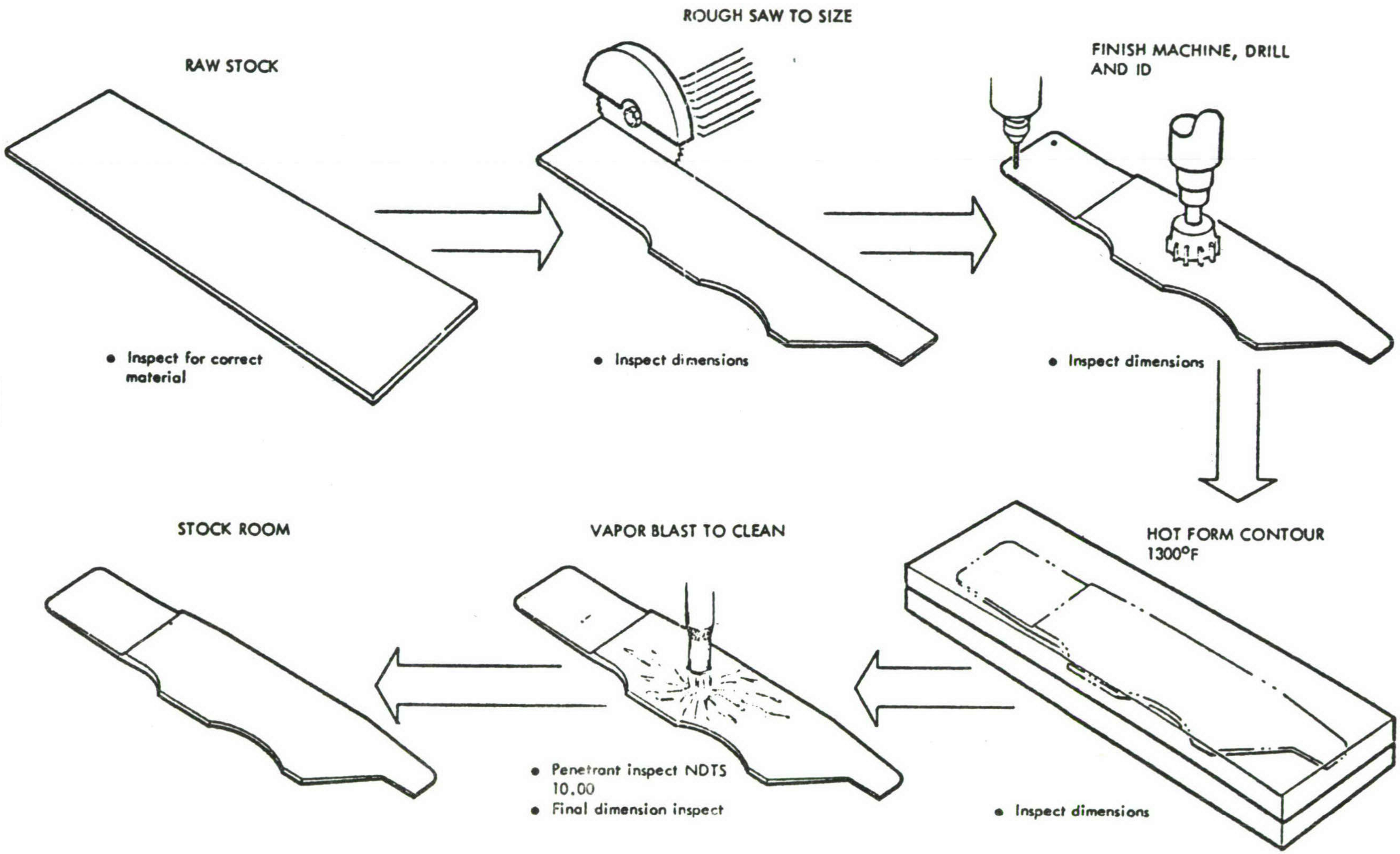


Figure 5 12W974-9/-10 DOUBLER, 6Al-4V TITANIUM (MIL-S-9046, CLASS #2)

F-111F WING MANUFACTURING SEQUENCE

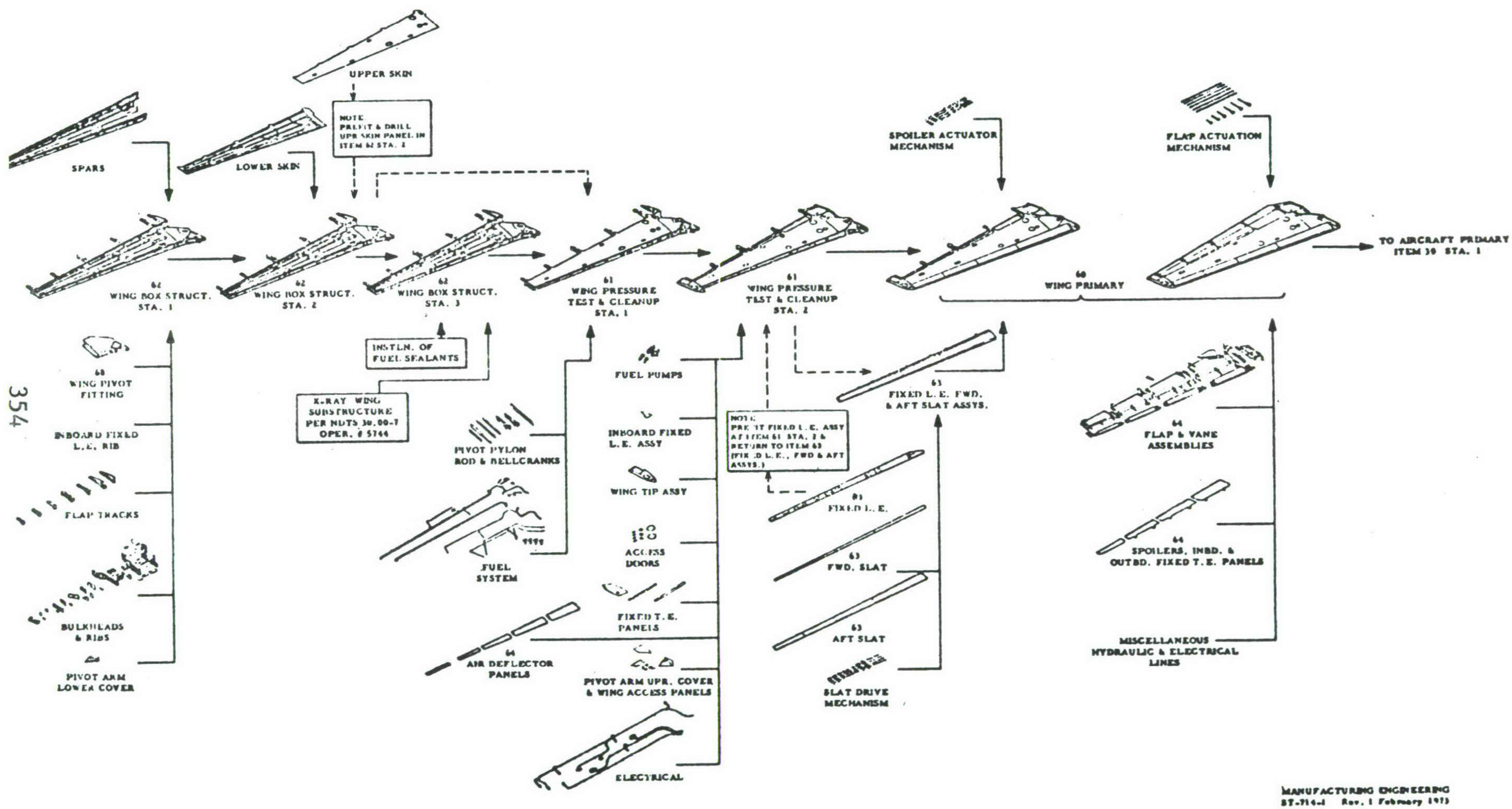


Figure 6

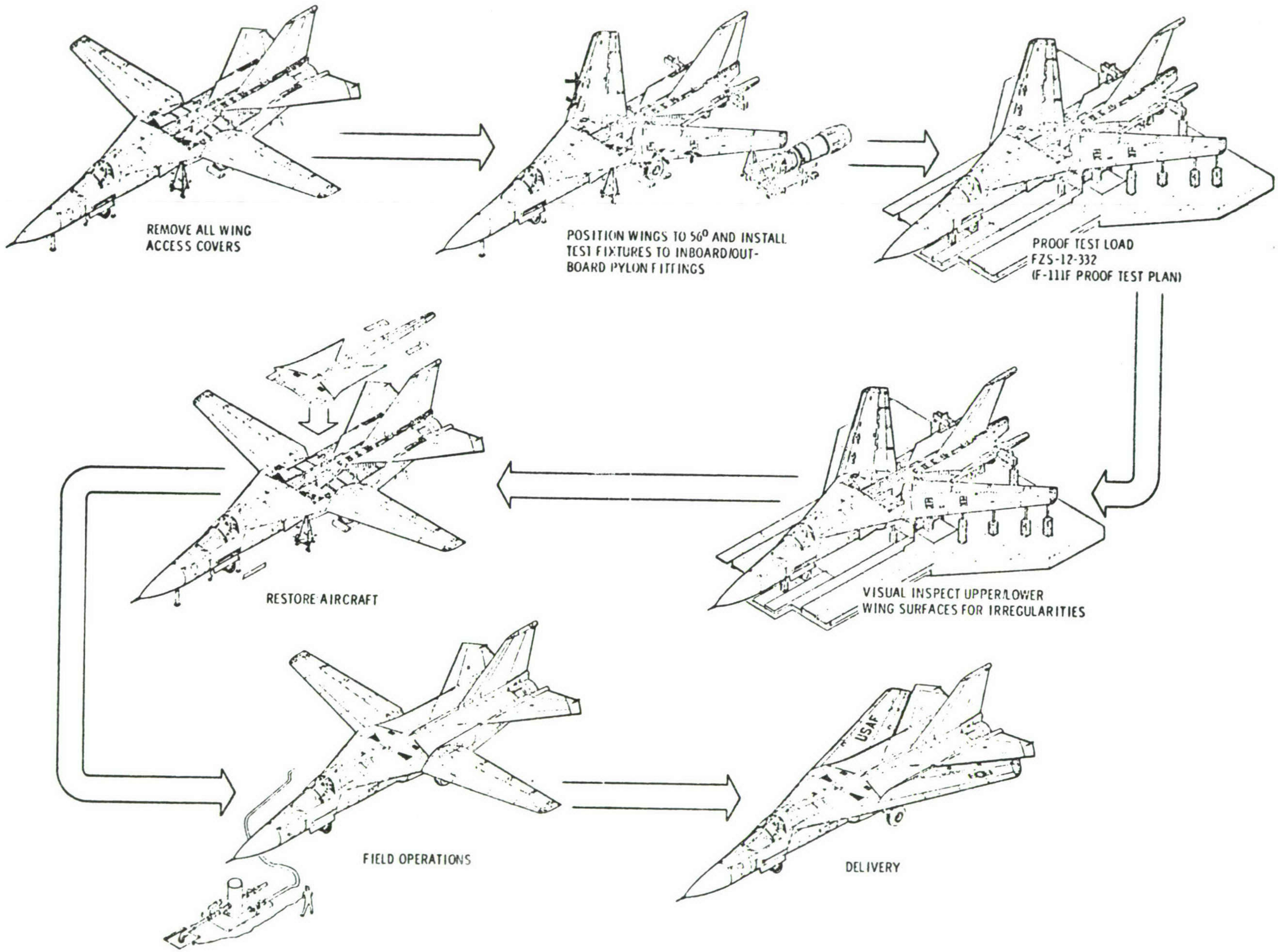


Figure 7 F-111F AIRCRAFT WING PROOF TEST, (COLD TEMPERATURE)
12AEI-11-1047B

Table 1 WING BOX STRUCTURE NDI EXPERIENCE SUMMARY

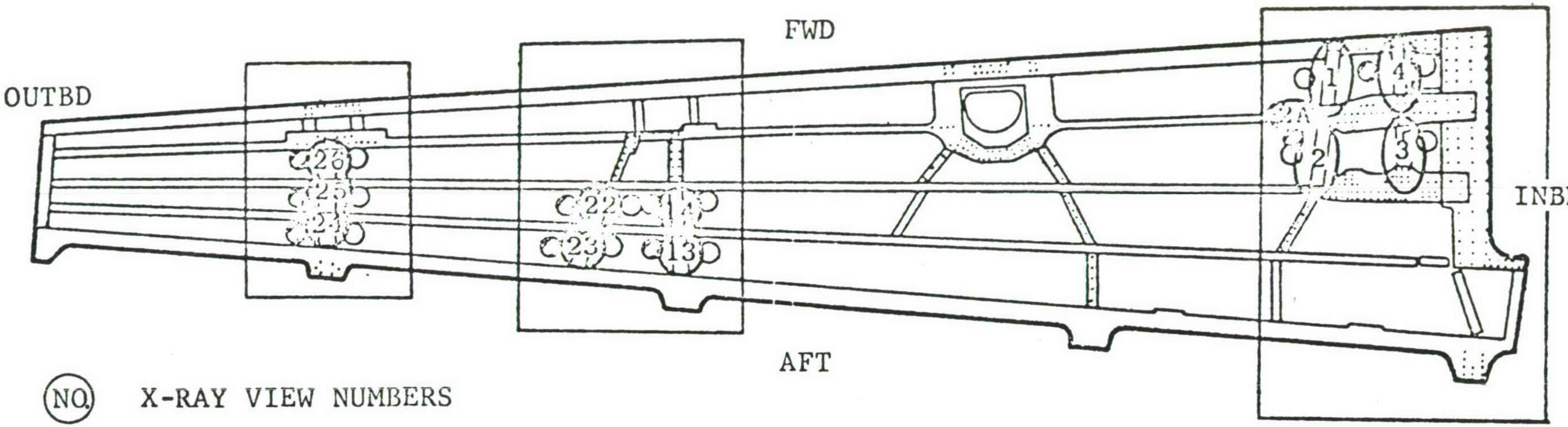
NDI PERFORMED	12W985 WEB SPAR	12W950 WING SKIN	12W915 BULKHEAD	12W908 WING SPAR	12W974 DOUBLER	ASSEMBLY WING BOX
Ultrasonic per NDTS 50.00				1 Time		
Rockwell Hardness per NDTS 15.00	1 Time	1 Time	1 Time	1 Time		
Penetrant Inspection per NDTS 10.00	1 Time	1 Time	1 Time	1 Time	1 Time	
X-ray Inspection per NDTS 30.00-7						1 Time

NOTE:

(1) No formal minimum flaw size established for F-111. AFML has opinioned: A clean crack, open to the surface, .030 inches in length by .003 inches in depth by .002 inches in width, for production. CA/FW Engineering has released FPS-1084, Zoning Standards and Procedures For Penetrant Method of Inspection, and M Standard # M501, which defines flaw criteria for penetrant inspection; however, these specifications are not applicable to the F-111 program.

(2) See Figure 8 for x-ray locations. Flaw size is 2% of material thickness for area defect and 50% of material thickness for tight crack.

(3) For flaw size criteria, see NDTS 50.00. (Dependent upon class of flaw)



(NO.) X-RAY VIEW NUMBERS

○ = FOCAL SPOT PLACEMENT, REGULAR SHOT

⊘ = FOCAL SPOT PLACEMENT, "A" SHOT

LEFT/HAND BOTTOM VIEWS

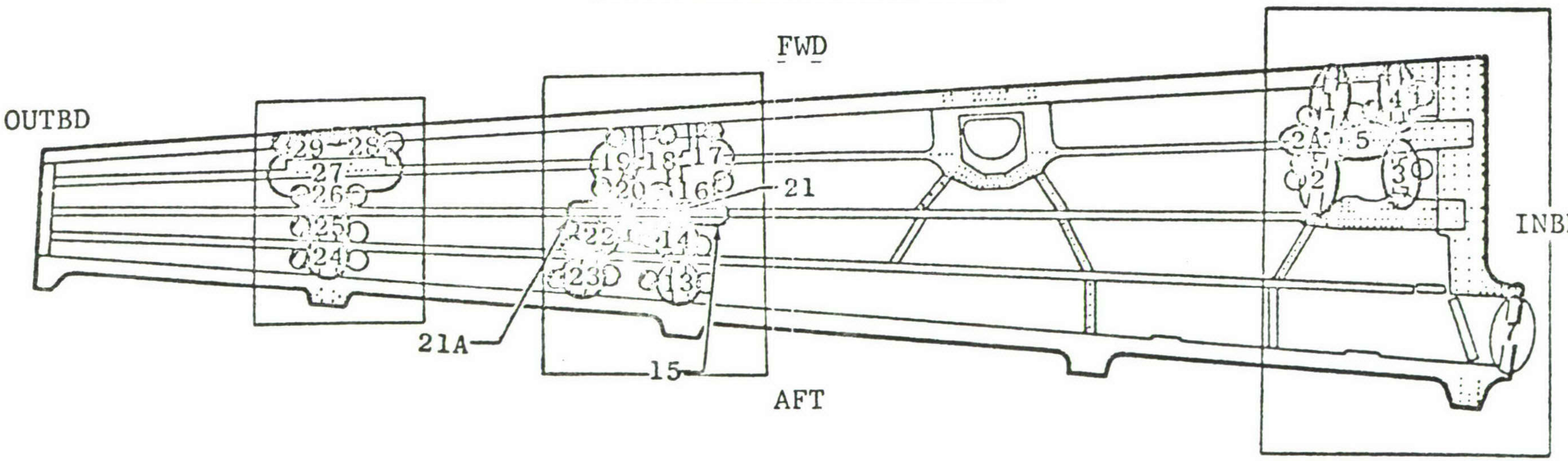


Figure 8 LEFT WING ILLUSTRATED. SAME VIEWS REQUIRED ON RIGHT WING

GENERAL DYNAMICS *Fort Worth Division* -NDTS- NONDESTRUCTIVE TEST STANDARD	NUMBER 10.00 ISSUE 2 SUPERSEDING 10.00 ISSUE 1 DATE 4 AUG 1972 PAGE 1 OF 18 PREPARED R. M. Walker PROCESS CONTROL J. E. Herr ENG'R E. R. [illegible] INSP. R. [illegible] AIR FORCE COORDINATED C. B. Schultz

LIQUID PENETRANT INSPECTION

**1.0 SCOPE

**1.0.1 This standard establishes specific process requirements, inspection procedures or techniques, and quality standards for penetrant inspection of metals.

*1.0.2 The inspection requirements referenced herein are applicable to parts or materials when conformance to the requirements of this standard is required by engineering drawing, procurement specification, purchase order (PO), contract, Nondestructive Test Standard (NDTS), etc., and as required in the manufacturing process to maintain quality standards.

1.1 SCHEDULING

1.1.1 Liquid penetrant inspection per this NDTS shall be employed on those raw stocks requiring (by contract, Engineering drawing, procurement specification, etc.), liquid penetrant inspection, AND:

(1) On castings, forgings and extrusions prior to and after machining to final dimensions.

(2) On all welds when specified by Engineering drawing requirement.

(3) After rework of defective welds.

(4) After severe cold working, stretch forming, straightening and heat treating, as specified.

1.2 AREA TO BE INSPECTED

1.2.1 For final inspection, all accessible surfaces are to be inspected. For "in process" utilization of liquid penetrant inspection, selected areas may be examined. In the inspection of weld zones, for example, only the weld areas are to be inspected.

2.0 SPECIFICATION CONFORMANCE

**2.0.1 This NDTS meets or exceeds the requirements of the documents listed below. The documents used when preparing this NDTS were those whose issue or revision letter is established by contract.

FPS-1001 - Castings, Engineering and Inspection Requirements for.

QADI Q-101 - Certification of Nondestructive Testing Personnel

FPS-0065 - Inspection and Acceptance Standards for Fusion Welds, Specification for

MIL-I-6866 - Inspection, Penetrant Method of

MIL-STD-410 - Qualification of Inspection Personnel

MIL-I-25135 - Inspection Material, Penetrant

FPS-0040 - Inspection Process Specification for Zoning Standards for Magnetic Particle

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3.0 EQUIPMENT REQUIREMENTS

3.0.1 The following equipment and accessories are required to perform liquid penetrant inspection in accordance with this procedure.

3.0.1.1 Liquid Penetrant inspection equipment (penetrant, rinsing, and developing tanks).

3.0.1.2 Darkened area or booth for black light examination.

3.0.1.2 One 125 foot-candle power (minimum) black light, or an adequate white light.

3.0.1.4 A supply of penetrants, emulsifiers and developers (Reference Table I).

The following groups of penetrants are authorized in accordance with USAF document MIL-I-25135:

** TABLE I

MIL-I-25135	MIL-I-6866	NAME
Group I	Type II Method C	Visible Dye Penetrant - Solvent Removable
Group II	Type II Method B	Visible Dye Penetrant - Post Emulsified
Group III	Type II Method A	Visible Dye Penetrant - Water Washable
Group IV	Type I Method A	Fluorescent Penetrant - Water Washable
Group V	Type I Method B	Fluorescent Penetrant - Postemulsified (Medium Sensitivity)
Group VI	Type I Method B	Fluorescent Penetrant - Postemulsified (High Sensitivity)
Group VII	Type I, Method C	Fluorescent Penetrant - Solvent Removable

NOTE: Aluminum and steel weld assemblies subject to contamination by use of Type I may be inspected using Type II materials.

Families of one sensitivity or manufactures shall not be mixed with that of another sensitivity or manufacturer, unless by an authorized letter of approval.

**3.0.1.5 A supply of coupons for testing penetrant sensitivity. Reference Magnaflux Part No. 14755 or equivalent.

4.0 PREINSPECTION PART PREPARATION

4.0.1 All items to be inspected shall be free from scale, grease, oily film, burrs and other coating or objects which may interfere with the application of the inspection process or test method.

4.0.1.1 Cleaning or deburring may be accomplished by one of the following methods:

CLEANING	DEBURRING
Vapor Degrease	Hand or Rotary Files
Solvents	Grinders
Descaling Solutions	Abrasive Blasting
Detergents	Sand Paper
Abrasive Blasting	Barrel Tumbling
Ultrasonic Cleaniner	

4.0.1.2 Other Methods may be employed as long as they do not:

(1) Leave a film on the surface that will interfere with the process.

(2) Produce an indication that can be readily interpreted as a discontinuity without extensive investigation and reprocessing.

(3) Close discontinuities by metal displacement.

4.0.1.3 Parts cleaned by any of the above methods except vapor degreasing or ultrasonic (if water is used) shall be hot water rinsed and thoroughly dried or vapor degreased prior to application of the inspection or test method.

4.0.1.4 When vapor degreasing is used with another method, vapor degreasing shall be the final step prior to inspection.

4.0.1.5 Discoloration due to low temperature heating of the metal does not affect the inspection process and need not be removed prior to inspection. For example 17-4 PH steel aged after final machining does not require vapor blasting before inspection to remove the discoloration.

4.0.1.6 When anodizing or similar surface finish is called out, inspection shall be accomplished prior to the processing operation, unless the part is machined or worked following the surface finish process, in which case the part shall be inspected again following the final machine or working operation. The finish need not be removed.

5.0 CALIBRATION PROCEDURE

5.0.1 Thermostat Control

5.0.1.1 The thermostat controlling the dryer cabinet shall be checked by Quality Control personnel at the beginning of each week.

5.0.1.2 The theremostat shall be set at 200°F and the dryer cabinet stabilized at this temperature for one (1) hour.

5.0.1.3 A thermometer, capable of measuring the tolerances stated in paragraph 6.0.5.4, shall then be used to check the temperature at the theremostat-element and at three other locations in the dryer cabinet. Temperature at these locations shall be within the tolerances specified in this NDTS.

5.0.1.4 A record log shall be maintained indicating the results of each check.

5.0.1.5 Black lights employed to review parts and assemblies shall emit radiation primarily between 3300 and 3900 angstrom units, and shall have a minimum intensity of 125 foot candles at 15 inches.

5.0.1.6 Black lights employed to review parts shall be checked and recorded every 90 days for intensity requirements established in paragraph 5.0.1.5. Lights not meeting these requirements shall be replaced. Intermediate inspection may be conducted at any time the light intensity is questioned.

5.0.1.7 Black lights employed as background lighting are not included in paragraph 5.0.1.5.

6.0 INSPECTION PROCEDURES (GENERAL APPLICATIONS)

6.0.1 For those part indexed in NDTS 10.00-0, refer to paragraph 6.1.

6.0.2 For all other parts/raw stocks requiring liquid penetrant inspection, the procedures contained under this paragraph (6.0) shall apply.

6.0.3 The surface condition shall conform to the requirements established in paragraph 4.0 of this procedure.

6.0.4 It is necessary that all parts be thoroughly dry after cleaning so that water or solvent will not hinder entrance of the penetrant into the discontinuity.

6.0.5 Application of Group III and Group IV (Method A, Types I & II).

6.0.5.1 Penetrant - Water-Washable.

(1) Apply penetrant solution to all surfaces to be inspected.

(2) Permit time specified in Table II for penetration.

(3) Caution should be exercised to prevent solution from puddling or collecting in pockets, radii or other areas.

(4) Fluorescent penetrant shall not be applied to parts whose temperature is in excess of 100^{o}F or lower than 60^{o}F.

6.0.5.2 Rinsing

(1) Rinsing shall be accomplished by the use of a nozzle sprayed tap water; temperature not to exceed 80 ± 20^{o}F.

(2) A minimum rinse time shall be used to remove only background color or fluorescence. Do not over rinse.

(3) Rinse under, or check, with black light to assure fluorescent penetrant removal.

6.0.5.3 Drying

(1) Drying, whenever possible, shall be accomplished in a recirculating hot air dryer that is thermostatically controlled and electrically heated.

(2) Temperature of the dryer shall be $200^{\circ}F \begin{array}{l} +25^{\circ}F. \\ -50^{\circ}F \end{array}$

(3) Parts should not be left in the dryer longer than is necessary to dry them. Excessive time may affect the sensitivity of the penetrant.

(4) When size or configuration of the part does not permit hot air drying, the parts may be air dried; however, the maximum processing time shall not exceed that permitted in Table III.

(5) All parts shall be reviewed as soon as possible after they are dry.

6.0.5.4 Dry Developer (when used)

(1) Prior to application of the dry developer, the part shall be thoroughly dried as specified in paragraph 6.0.5.3.

(2) The dry developer shall be applied with a hand powder bulb, powder gun, soft brush or other suitable method.

(3) The excess powder may be removed by shaking and tapping on the part gently or by blowing with low pressure compressed air that is clean and dry.

(4) Developing time prior to review shall be as specified in Table III.

6.0.5.5 Wet Developer

(1) Prior to application of the wet developer the part should be relatively dry.

(2) Wet developer shall be applied by spraying, dipping or flowing.

(3) The part should be allowed to drain a few minutes prior to placing in a recirculating hot air dryer.

(4) When suspension type developer powder is used in water it must be agitated periodically to keep it from settling to the bottom. When water soluble powder is used agitation is not required.

6.0.5.6 Inspection

(1) After the parts have thoroughly dried they should be inspected as soon as possible. Inspectors shall allow sufficient time for vision adjustment prior to evaluation.

(2) Review of groups IV, V, VI and VII (fluorescent) shall be conducted under black light in a darkened area.

(3) Review of groups I, II and III shall be conducted under adequate white light.

(4) Review shall be conducted in accordance with this procedure or, specific NDTS if prepared.

6.0.6 Application of Group II, V, and VI (Method B, Types I and II).

6.0.6.1 Penetrant - Postemulsified (paragraph 6.0.8).

(1) The postemulsified penetrants differ from water washable in that an emulsifier is required to make the penetrants water washable.

(2) The emulsifier is applied at the end of the penetration period and prior to water washing.

(3) The emulsification time is critical and must be observed within the limits specified in Table III. When exceeded the part shall be thoroughly washed, dried and reprocessed through penetrant.

6.0.7 Application of Groups I and VII (Method C, Types I and II)

6.0.7.1 Penetrant - Solvent Removable (Para. 6.0.8).

(1) The solvent removable penetrants differ from postemulsified penetrants in that a water wash is not required.

(2) Solvent is applied at the end of the penetration period and removed with cheesecloth or other suitable lint free cloth.

6.0.8 Penetrant Processing Sequence.

Groups III,IV,V & VI Type I and II Method A (Water Washable)		Groups II, V & VI Type I and II Method B (Postemulsified)		Groups I & VII Type I and II Method C (Solvent Removable)
Dry or Nonaqueous Developer	Wet Developer	Dry or Nonaqueous Developer	Wet Developer	Nonaqueous Developer
Apply Penetrant	Apply Penetrant	Apply Penetrant Drain	Apply Penetrant Drain	Apply Penetrant
Drain	Drain	Apply Emulsifier	Apply Emulsfier	Drain
Water Wash	Water Wash	Rinse	Rinse	Apply Remover
Dry	Apply Wet Developer	Dry	Apply Developer	Apply Developer
Apply Developer	Dry	Apply Developer	Dry	Inspect
Inspect	Inspect	INSPECT	INSPECT	

TABLE II

MATERIAL	FORM	TYPE OF DISCONTINUITY	WATER WASHABLE MEDIUM & HIGH SENSITIVITY PENETRATION TIME (MINUTES)	WATER WASHABLE LOW SENSITIVITY PENETRATION TIME (MINUTES)	POST EMULSIFIED PENETRATION TIME (MINUTES)
Aluminum	Castings	Porosity	3-5	5 to 15	5
		Cold Shuts	3-5	5 to 15	5
	Extrustions & forgings	Laps	3-8		10
	Welds	Lack of fusion	5-8	30	5
		Porosity	5-8	30	5
	All	Cracks	5-8	30	10
	All	Fatigue cracks	3-8		30
Magnesium	Castings	Porosity	3-5	15	5
		Cold Shuts	3-5	15	5
	Extrusions & forgings	Laps	3-8		10
	Welds	Lack of fusion	5-8	30	10
		Porosity	5-8	30	
	All	Cracks	5-8	30	10
	All	Fatigue cracks	3-8		30
Steel	Castings	Porosity	5-8	30	10
		Cold Shuts	5-8	30	10
	Extrusions & forgings	Laps	3-8		10
	Welds	Lack of fusion	5-10	60	20
		Porosity	5-10	60	20
	All	Cracks	5-8	30	20
	All	Fatigue cracks	3-8		30
Brass and Bronze	Castings	Porosity	3-5	10	5
		Cold Shuts	3-5	10	5
	Extrusions & forgings	Laps	3-8		10
	Brazed parts	Lack of fusion	3-5	15	10
		Porosity	3-5	15	10
	All	Cracks	5-8	30	10
Plastics	All	Cracks	3-8	5 to 30	5
Glass	All	Cracks	3-8	5 to 30	5
Carbide-tipped tools		Lack of fusion	5-8	30	5
		Porosity	5-8	30	5
		Cracks	5-8	30	20
Titanium and high-temperature alloys	All	All			20 to 30
All metals	All	Stress or intergranular corrosion	3-8		240

NOTE: Emulsion time is 30 seconds to 5 minutes. Developing time is one-half of penetration time.

TABLE III

GROUP	EMULSIFICATION TIME (MINUTES)		DEVELOPING TIME (MINUTES)	REVIEW TIME (MINUTES)	TOTAL TIME (HRS)
	MIN	MAX		MAX	MAX
IV, V, VI,VII	1	5	120(2)	30	1.40 3.00
I, II, III	1	5	5	10	0:50

(1) For oven dried parts, 1/2 penetration time.
(2) For parts dried in still air at 70°F.

6.0.9 Only penetrant solutions listed in USAF document, MIL-I-25135 or approved by letters of authorization from Wright Patterson Air Defense Command shall be used by Convair Aerospace Division, Fort Worth. Deviations under unique conditions must be submitted to Process Control Metallurgical section for approval before use.

6.0.10 The penetrant inspector shall once each week process a coupon equivalent to that illustrated in Figure 1, Magnaflux Part No. 14755.

6.0.10.1 The test coupon shall be cleaned by a vigorous scrubbing with bristle brush and liquid solvent followed by a vapor degrease. Any evidence of retained contamination shall be cause for repeated cleaning or rejection of the coupon.

6.0.10.1.1 <u>Cleaning Test Coupon for Re-Use</u> - Before the Test Coupon is used again for a comparison test, it shall be heated slowly with a gas burner to 800°F, as determined by an 800°F Tempilstik, or equal, after which the test coupon shall be quenched in cold water. It shall then be heated at 225° ± 5°F for 15 minutes to drive off any moisture in the cracks and allowed to cool to room temperature.

6.0.10.2 One end of the test coupon shall be processed with solutions in the production system. The other end shall be processed in new solutions.

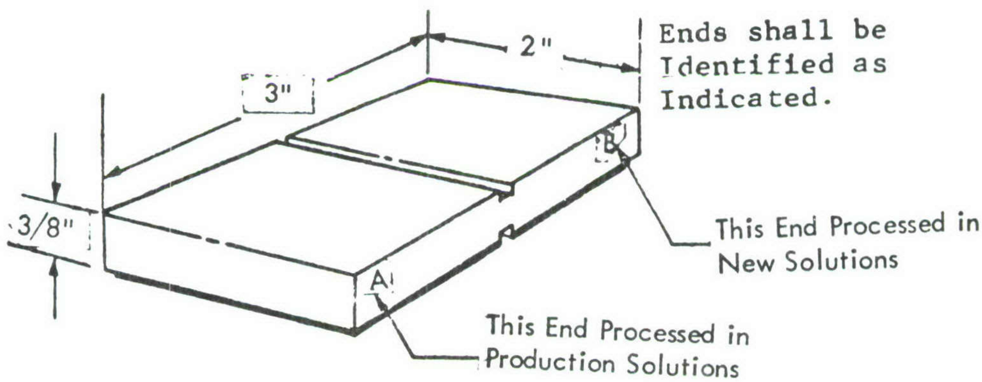
6.0.10.3 Processing shall conform to the applicable portion of paragraph 6.0.8.

6.0.10.4 The sensitivity of the two halves shall compare favorably.

6.0.10.5 The record shall then be stamped and the results posted as satisfactory or unsatisfactory. If questions arise or the test is unsatisfactory, Process Control Metallurgical section shall be notified immediately.

6.0.10.6 Process Control shall take immediate corrective action. By determining which solution is out of control, and contact Maintenance for replacement of solution.

6.0.10.7 Aerosol systems are not included in this test since the solutions are not reused.



Typical of Sensitivity Test Coupons

Figure I

6.0.11 Table IV should be a guide to determine which solution is out of control.

W242399

TABLE IV

MATERIAL		TEST EQUIPMENT
Penetrant Oil	(a)	Check fluorescence under a black light.
Emulsifier	(a)	Water contamination: a sample drawn from the questioned system shall tolerate the addition of at least 15% of water without separating, clouding, thickening or jelling.
	(b)	The viscosity should be between 300-200 centistokes at room temperature.
	(c)	The material shall remove surface penetrant sufficiently to eliminate background fluorescence with not more than five (5) minutes emulsification time.
Developer	(a)	Specific gravity should be between 1.002 and 1.070
	(b)	The material shall exhibit no fluorescent properties.

*6.0.12 Water base developer, after mixing, shall be tested with a hydrometer in a sample taken from the hose or from dipping deep in the tank as applicable, to assure that the concentration is within the manufacturer's recommended range.

*6.0.13 Water-washable test shall be conducted in accordance with MIL-I-25135. This test shall be accomplished at least once a month or before replenishing the materials in the tanks which ever occurs first. The comparisons may be made with a sample of the same penetrant, emulsifier or developer batch, which has been set aside in a closed container for testing purposes.

6.1 INSPECTION PROCEDURES (SPECIAL APPLICATION)

6.1.1 For those parts indexed in NDTS 10.00-0 the procedures contained under this paragraph (6.1) shall apply.

6.1.2 For all other parts and raw stock requiring liquid penetrant inspection, refer to paragraph 6.0.

6.1.3 Specific instructions will indicate techniques and acceptance criteria to be used in evaluation of specific parts or assemblies when designated by Engineering requirement.

7.0 DEFECT INDICATION EVALUATION

7.0.1 It shall be the full responsibility of the inspector to determine whether or not indications are true or false, prior to rejection of the part or assembly.

7.0.2 Indications within the limits of the applicable NDTS or Engineering Specification or other applicable document shall be considered acceptable without investigation.

7.0.3 Indications not within the limits of the applicable control specification shall be investigated to determine, as near as possible, their actual size, shape, location and orientation.

7.0.3.1 Indications may be investigated by any method except by scraping or gouging. Those methods which are acceptable, include, files - both hand and rotary, emery, grinders, etc.

NOTE: Caution should be exercised to use an instrument coarse enough to cut and not smear, and at the same time it must be fine enough to not scrape or gouge the part or produce deep scratches. A good general purpose grit is 240.

7.0.4 Consideration shall be given to the area in which the indication is located, as well as, quantity, type, size, and orientation.

7.0.4.1 Discontinuities totally within an area which will be removed by subsequent processing, such as, machining, grinding, shall not be cause for rejection.

7.0.4.2 Irregularities such as shallow cracks, laps, certain mold imperfections, etc., are common to cast surfaces (castings and welds) and should be investigated by removing some of the surface, not to exceed .010 inch or minimum B/P tolerance.

7.0.4.3 Irregularities such as shrinkage, porosity, inclusions, etc., are common to machined and ground surfaces of welds and castings. They are not considered detrimental and should not be rejected unless:

(1) They display dimensions in excess of that permitted by the applicable specification or, radiographic standard.

(2) Surface imperfections are not permitted because of subsequent processing such as plating, where the discontinuity will not be bridged by the finish, thus, leaving a discontinuity.

8.0 ACCEPT-REJECT CRITERIA

8.0.1 Parts or assemblies that meet the requirements of the applicable specification shall be considered acceptable.

8.0.2 Parts or assemblies that do not meet the applicable specification requirements shall be considered rejectable and dispositioned in accordance with QADI J-101 of current issue.

8.0.3 The following items shall be considered rejectable unless allowed by applicahle specification or other documentation:

(1) Cracks
(2) Laps
(3) Seams
(4) Misruns
(5) Shrink Cavities
(6) Cold Shuts
(7) Discontinuities not normally considered rejectable but aligned to cause excessive stress concentrations.

9.0 POST INSPECTION REQUIREMENTS

9.0.1 Quality Control personnel shall be responsible for maintaining adequate records to the requirements of this procedure.

9.0.2 Rejection paperwork shall identify defect location and dimension.

9.0.3 All traces of liquid penetrant materials used in this inspection shall be removed from parts following evaluation by inspection personnel.

9.0.4 Accepted parts shall be stamped in accordance with applicable QADI instructions.

*9.0.5 Control Check Sheets shall be maintained at penetrant inspection area. (Ref. pages 17 & 18).

W24 2403

GENERAL DYNAMICS
Convair Aerospace Division
Fort Worth Operation

QUALITY ASSURANCE PROVISIONS
PENETRANT INSPECTION

DATE:__________

QUALITY ASSURANCE CHECK	PROCEDURE	SATIS-FACTORY	UNSAT-ISFACTORY	STAMP
SENSITIVITY OF MATERIALS AND PROCESS 1. TANK NO.____ 2. TANK NO.____ 3. TANK NO.____ 4. TANK NO.____	THIS CHECK SHALL DETERMINE THE ABILITY OF THE INSPECTION MATERIALS TO DETECT SURFACE DISCONTINUITIES. THE TEST PENETRANT FAMILY SHALL BE APPLIED TO ONE HALF OF THE TEST BLOCK IN ACCORDANCE WITH NDTS 10.00 AND THE REFERENCE PENETRANT FAMILY SHALL BE APPLIED TO THE REMAINING HALF OF THE BLOCK. THE SENSITIVITY OF THE TWO HALVES SHALL COMPARE FAVORABLY.			1. 2. 3. 4.
SPECIFIC GRAVITY AQUEOUS WET DEVELOPER	THIS CHECK SHALL ASSURE THAT THE DEVELOPER CONCENTRATION IS WITHIN THE MANUFACTURER'S RECOMMENDED RANGE. A SAMPLE OF THE DEVELOPER SOLUTION SHALL BE TESTED WITH A HYDROMETER TO DETERMINE THE SPECIFIC GRAVITY AND MAINTAIN THE RECOMMENDED CONCENTRATION. THIS CHECK SHALL BE PERFORMED TWICE WEEKLY. DATE | READING			
DEVELOPER CONTAMINATION	THIS CHECK SHALL DETERMINE CONTAMINATION (FLUORESCENCE) OF BOTH WET AND DRY DEVELOPERS. THE DEVELOPER SHALL BE CHECKED WITH A BLACK LIGHT. ANY DEVELOPER FLUORESCENCE IS UNACCEPTABLE AND REQUIRES REPLACEMENT. DRY DEVELOPER SHALL BE CHECKED FOR DISCOLORATION OR AGGLOMERATION. WET DEVELOPER SHALL BE CHECKED FOR SETTLING OF SOLIDS, SCUM ON SURFACE AND INABILITY TO WET THE SURFACE BEING INSPECTED.			
ULTRAVIOLET (BLACK LIGHT) INTENSITIES	THIS CHECK SHALL ASSURE BLACK INTENSITY REQUIREMENTS ARE MAINTAINED. INTENSITIES SHALL BE DETERMINED WITH A LIGHT METER. INTENSITY SHALL BE 125 FOOT CANDLES MINIMUM IN THE CENTER OF THE BEAM, 15 INCHES FROM FACE OF FILTER. BL NO. | INTENSITY			
WHITE LIGHT INTENSITIES	THIS CHECK SHALL ASSURE WHITE LIGHT INTENSITY REQUIREMENTS FOR INSPECTION WITH NON-FLUORESCENT METHODS ARE MAINTAINED. INTENSITIES SHALL BE DETERMINED WITH A LIGHT METER. INTENSITY SHALL BE 100 FOOT CANDLES MINIMUM AT THE NORMAL WORKING HEIGHT ON SURFACE OF PARTS BEING EXAMINED. INTENSITY READING COL. 26-D____			
DRYER CABINET THERMOSTAT CONTROL	THIS CHECK SHALL ASSURE PROPER TEMPERATURE CONTROL OF THE DRYER CABINETS. THE TEMPERATURE OF THE DRYER SHALL BE $200^oF \pm {}^{25^oF}_{50^oF}$. MEASUREMENTS SHALL BE TAKEN AT THE THERMOSTAT-ELEMENT AND THREE OTHER LOCATIONS IN THE DRYER CABINET. CABINET NO. | TEMPERATURE			

-WEEKLY-

W 2 4 2 4 0 4

GENERAL DYNAMICS
Convair Aerospace Division
Fort Worth Operation

QUALITY ASSURANCE PROVISIONS
PENETRANT INSPECTION

DATE:________________

QUALITY ASSURANCE CHECK	PROCEDURE	TEST TAG NO.	SATIS-FACTORY	UNSATIS-FACTORY	STAMP
WATER WASHABILITY	PROCESS CONTROL FUNCTION: THE INTENT OF THIS CHECK IS TO ASSURE ADEQUATE REMOVABILITY OF PENETRANTS. PREPARE PRODUCTION SAMPLE OF PENETRANTS AND EMULSIFIERS FOR PROCESS CONTROL EVALUATION VIA TEST TAG.				
TANK NO.______					
TANK NO.______					
TANK NO.______					
TANK NO.______					
WATER TOLERANCE	PROCESS CONTROL FUNCTION: THE INTENT OF THIS CHECK IS TO ASSURE THAT PENETRANTS AND EMULSIFIERS USED IN OPEN TANKS ARE WITHIN THE WATER TOLERANCE RANGE. PREPARE PRODUCTION SAMPLES OF PENETRANTS AND EMULSIFIERS FOR PROCESS CONTROL EVALUATION VIA TEST TAG.				
TANK NO.______					
TANK NO.______					
TANK NO.______					
TANK NO.______					
COMPARISON TEST PENETRANT	PROCESS CONTROL FUNCTION: THE INTENT OF THIS CHECK IS TO ASSURE THE QUALITY OF IN-USE PENETRANTS. THE SAMPLES SHALL BE VISUALLY EXAMINED FOR PRECIPITATION, SEPARATION, LOSS OF FLUORESCENT BRIGHTNESS. PREPARE PRODUCTION SAMPLES OF PENETRANTS FOR PROCESS CONTROL EVALUATION VIA TEST TAG.				
TANK NO.______					
TANK NO.______					
TANK NO.______					
TANK NO.______					

- MONTHLY -

W24 2405

NONDESTRUCTIVE TEST STANDARD

RELEASE ORDER

RELEASE GROUP DWR

NDTS NO.	ISSUE NO.	AMEND NO.	REV. LTR.	DATE
10.00	2			JAN 25 1973

APPLICABLE TO PART NO.

GENERAL *

NDTS TITLE:

LIQUID PENETRANT INSPECTION

AUTHORIZATION (S.O.)

DOCUMENT EFFECTIVITY		PLANNING EFFECTIVITY	
TYPE VERSION	EFFECTIVITY	GDFW	VENDOR
	**		

REASON FOR CHANGE:

REMARKS:

* General requirement for Penetrant Inspection.
** Record change.

PREPARED BY:	DATE: 9/26/72	PQA (When vendor affected)	DATE:
PROCESS CONTROL SUPERVISOR:	DATE: 9/	AUTHORIZED BY: 377	DATE:

DEPARTMENT 273

GENERAL DYNAMICS Fort Worth Division -NDTS- NONDESTRUCTIVE TEST STANDARD	NUMBER 15.00 ISSUE 1 SUPERSEDING ______ ISSUE ______ DATE 28 Jun. 1971 PAGE 1 OF 19 PREPARED [signature] PROCESS CONTROL [signature] ENG'R [signature] INSP. [signature] AIR FORCE [signature]

HARDNESS TESTING, METHOD OF INSPECTION

1.0 SCOPE

1.0.1 This document establishes procedures to be used when hardness testing of metals to assure correct heat treat methods have been applied to materials used in the fabrication and installation of aircraft components.

1.0.2 This procedure applies only to approved methods and equipment for determining the hardness of materials to Engineering requirements.

1.1 SCHEDULING

1.1.1 Hardness testing shall be accomplished on the following materials as specified:

(1) Steel - All steel except 300 series.

(2) Aluminum - Hardness testing shall be accomplished on aluminum only to differentiate between annealed and heat treated material and shall not be converted to tensile properties.

(3) Magnesium - Hardness testing is not valid and shall not be accomplished.

(4) Titanium - Hardness testing shall be accomplished only when the hardness is specified.

(5) Brass - Hardness testing shall be accomplished only when the hardness is specified.

1.1.2 The following methods are approved for hardness testing at or for the Fort Worth Division. Reference Table I.

TABLE I

APPROVED METHODS

METHOD	STATIONARY	PORTABLE
ROCKWELL	NORMAL & SUPERFICIAL	NORMAL
BRINELL	500, 1500, 3000 Kg LOAD	500 Kg LOAD

1.1.3 The thickness-hardness relationships established in Figures I and II are intended to be a guide but should be followed as closely as possible. Thinner or softer materials may be tested, <u>but, in no case is a hardness reading valid if the impression is visible on the opposite side of the material</u>.

1.1.4 Readings are permitted on the various scales within the range specified in Table II.

1.1.5 Conductivity Tester/Webster Pliers

1.1.5.1 Conductivity tester and webster pliers may be used as a comparator when checking hardness on aluminum alloys.

1.1.5.2 Conductivity tester and/or webster pliers hardness readings shall be compared against actual hardness readings obtained on stationary Rockwell equipment from a production part.

TABLE II

WORKABLE RANGE OF THE VARIOUS SCALES

SCALE SYMBOL	PENETRATOR	LOAD Kg	DIAL COLOR	WORKABLE SCALE RANGE AND APPLICATIONS
		ROCKWELL NORMAL		
A	DIAMOND CONE	60	BLACK	A 60 TO A 86 - FOR EXTREMELY HARD MATERIALS WHICH MIGHT CHIP THE INDENTER UNDER HIGHER LOADS (TUNGSTEN CARBIDE, ETC.) AND FOR HARD STEEL SHEET TOO THIN FOR HEAVIER LOADS. REFERENCE FIGURE I
D		100		D40 to D77 - FOR THIN SHEET AND MEDIUM CASE HARDENED STEEL.
C		150		C20 - C70 - FOR STEEL, HARD CASTINGS, TITANIUM, DEEP CASE HARDENED STEEL AND IN GENERAL, MATERIALS HARDER THAN B100
B	1/16" DIA BALL	100	RED	B0 TO B100 - FOR COPPER, SOFT STEEL, ALUMINUM, ETC.
E	1/8" DIA. BALL	100	RED	E57 TO E100 - ALUMINUM AND MAGNESIUM ALLOYS AND BEARING METALS.
		ROCKWELL SUPERFICIAL		
15N 30N 45N	DIAMOND CONE	15 30 45	BLACK	
15T 30T 45T	1/16" DIA. BALL	15 30 45	RED	
		BRINELL		
NOTE 1		500 1500 3000		BHN 26 TO BHN 170 BHN 48 TO BHN 300 BHN 100 to BHN 770

NOTE 1: EITHER STANDARD STEEL OR TUNGSTEN CARBIDE BALLS MAY BE USED FOR BRINELL HARDNESS VALUES UP TO BHN 450. THE TUNGSTEN CARBIDE BALL SHALL BE USED FOR VALUES ABOVE BHN 450.

2.0 SPECIFICATION CONFORMANCE

2.0.1 The procedures defined in this NDTS require compliance with the contractual requirements of the following specification:

Federal Test Method Std. No. 151 - Metals; Test Methods.

ASTM E 18-67 - Standard Methods of Test for Hardness

2.0.2 Vendor laboratories performing hardness testing on materials and/or parts must be approved by Convair Aerospace Division, Fort Worth, Texas, prior to furnishing any nondestructive test services controlled by this standard.

3.0 EQUIPMENT REQUIREMENTS

3.0.1 The following equipment and accessories are required to perform hardness testing inspection in accordance with this procedure.

3.0.1.1 Stationary or portable Wilson hardness tester (150 KG Load) or equivalent.

3.0.1.2 Stationary or portable Brinell hardness tester (3000 KG Load) or equivalent.

3.0.1.3 Stationary or portable superficial hardness tester, conductivity meter (Magnaflux FM-100), Webster pliers, or equivalent.

3.0.1.4 A supply of anvils for the positioning of work on the machines.

3.0.1.5 A supply of penetrators for the material to be tested and adaptable to the equipment being used.

3.0.1.6 A supply of test blocks in the heat treat ranges for which the machines will be used,(Rockwell) or equivalent.

4.0 PREINSPECTION PART PREPARATION

4.0.1 All surfaces to be inspected shall be clean, free from scale, plating, paint and other finishes or surface conditions that will interfere with the operation. Mill decarb permitted by material specifications shall be removed prior to hardness testing.

4.0.2 All surfaces to be inspected shall have either a fine machine, or a 240 grit surface, or better.

4.0.3 Flat surfaces shall be within 5 degrees of parallel.

4.0.4 Both surfaces of the piece to be tested, and the anvil surface, shall be prepared to prevent cushioning or springback.

4.0.5 The work and tester must be positioned and adequately supported so that neither will slide, roll, sway, rock, or be subjected to any movement or excessive vibration while the test is being conducted.

4.0.6 The proper scale, weights, anvil, indentor, etc., must be chosen, reference Table II. If the material is too hard for the scale chosen, the penetrator may be damaged or the test will be insensitive. If the material is too soft, readings will be erratic or off the scale. The penetrator shall be checked every 90 days for damage. Maintain record of checks at equipment location.

4.0.7 Test impressions for both Rockwell and Brinell shall be separated by at least 2 1/2 diameters, and at least 2 1/2 diameters from the edge of the material.

5.0 CALIBRATION PROCEDURES

5.0.1 Calibration as defined by DSP 9-27.1 is not required.

5.0.1.1 Accuracy of the hardness testing equipment shall be made in the following manner.

5.0.1.1.1 Equipment accuracy check shall be in accordance with ASTM E 18-67 and the requirements of this procedure.

5.0.1.1.2 Accuracy of the hardness testing equipment shall be checked using the standardized test block method.

5.0.1.2 Verification of hardness testers shall be verified for the scales and ranges using test blocks to the accuracy specified in Table III.

TABLE III

SCALE OR Kg LOAD	LOW		MEDIUM		HIGH	
	RANGE	ACCURACY[1]	RANGE	ACCURACY[1]	RANGE	ACCURACY[1]
ROCKWELL						
C	20 to 30	1.0	35 to 55	1.0	59 to 65	0.5
B	40 to 59	1.5	60 to 79	1.0	80 to 100	1.0
30N	40 to 50	1.0	55 to 73	1.0	75 to 80	1.0
30T	43 to 56	1.0	57 to 70	1.0	71 to 82	1.0
BRINELL						
500	30 to 70	3%[2]	77 to 120	3%[2]	125 to 160	3%[2]
1500	50 to 130	3%[2]	135 to 210	3%[2]	220 to 290	3%[2]
3000	100 to 300	3%[2]	350 to 500	3%[2]	600 to 750	3%[2]

[1] The mean of three readings on the test block shall not vary from the stated hardness of the test block by more than the values in Table III.

[2] Average diameter measured in two (2) directions 90° apart from three (3) indentions.

5.0.1.3 Accuracy of stationary testers shall be tested prior to first production use during a 24 hour test, and as necessary thereafter. If not used on a shift, no test is required. See Note 1. Maintain daily record of accuracy checks at equipment location.

5.0.1.4 The test shall be made by selecting the proper scale, penetrator, weight(s), anvil and a test block within two points of the acceptable hardness range for the material being tested.

5.0.1.5 Make sure all surfaces are free from dust, pits or other objects or blemishes.

5.0.1.6 Make three impressions in the block. They must be within the limits specified in Table III.

5.0.1.7 Test blocks for Rockwell shall be used on <u>one side only</u>. Test blocks for Brinell may be used on two sides - they <u>must be adjacent</u> and <u>not opposite sides</u>.

5.0.1.8 Test blocks <u>shall not</u> be reground and used again. When surface area has been sufficiently used up the block shall be discarded.

5.0.1.9 Any tester showing evidence of malfunction or that is not within the accuracy requirements of Table III shall be tagged and not used until the necessary repairs have been made and the tester recalibrated to the requirements of ASTM E10-64 for Brinell machines or E18-67 for Rockwell machines.

5.0.1.10 Unless otherwise specified by Fort Worth Division Process Control, in writing, all steel parts shall be hardness tested after heat treatment prior to final acceptance.

5.0.1.11 Hardness testing of 2024 aluminum shall be accomplished following solution heat treating but prior to aging. Other heat treatable aluminum shall be hardness tested after heat treatment to final B/P condition.

5.0.1.12 Hardness testing equipment shall be located in areas away from grinding dust, heat treat furnaces, severely vibrating equipment, such as drop hammers, etc.

5.0.1.13 When shock or vibration affect the operation of the equipment or the validity of the reading, shock mounts shall be installed on all testers in the area.

5.0.1.14 All testers shall be provided with dust covers, and when the equipment is not used for a period of 4 hours or more the cover shall be installed.

NOTE: Accuracy of portable testers shall be checked prior to use. The hardness test block used shall be within two (2) points of the hardness range to which the production parts will be tested.

6.0 INSPECTION PROCEDURES (GENERAL APPLICATIONS)

6.0.1 Rockwell Hardness Test

6.0.2 The Rockwell hardness test is a differential depth measurement. The Rockwell hardness number (RHN) is a measurement of increments of depth as illustrated in Figure I. Each RHN represents 80 millionths of an inch indentation for Rockwell normal and 40 millionths of an inch for Rockwell superficial.

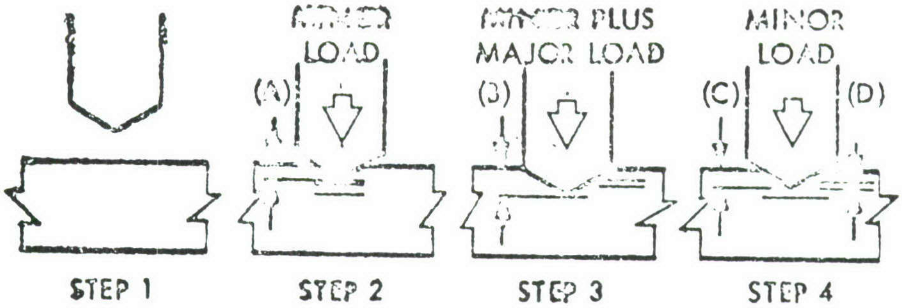


FIGURE I

The only difference between Rockwell normal and superficial is the applied loads. The minor load is 3 kilograms for superficial and 10 kilograms for normal, and the major loads are specified in Table II.

Step 1: After the proper scale, anvil, weight(s), indenter, etc., have been selected and the part placed on the anvil, or for portable work, the tester has been adequately positioned on the part and securely clamped, the following procedure will be followed:

Step 2: Apply the minor load by slowly elevating the specimen and anvil to the indenter until the proper load has been applied. This is normally when the small hand points to a dot, triangle or other mark and the large hand is pointing to a specified area on the face of the scale. The dial scale is now turned to the "Set" or zero position. The indenter has penetrated the material to a depth "A" as illustrated in Figure I, Step 2.

NOTE: The minor load is very carefully controlled and is of vital importance - the penetrator is firmly seated in the material, below minor surface imperfections and surface unevenness and establishes a definite zero point for the rest of the test. Jerky or uneven elevation of the test specimen when applying the minor load can cause

several points error in the final reading. Considerable care is required during this part of the test.

Step 3: Apply the major load by pressing the trip bar. The total load, major load plus minor load is now acting on the penetrator. The load lever should not be forced. The penetrator has now penetrated to depth "B" Figure I, Step 3.

Step 4: Remove the major load by returning the load lever to its starting position. This must be within 2 seconds after the motion of the lever has stopped, or as soon as the large hand stops moving. The minor load only is acting on the penetrator so that spring in the machine or work (not due to setup) was compensated for when the dial was zeroed in Step 2. The penetrator is now resting as illustrated in Figure I, Step 4, at the depth "C".

Step 5: The reading is taken, with the minor load acting but not the major load, and shall be designated to 1/2 of a division. The RHN is the depth "D" which is the difference between depths "A" and "B" and is measured by direct reading of the proper scale.

6.0.3 If readings are erratic, consideration should be given to (1) surface condition, (2) proper weights, scale, anvil, etc., (3) steadiness of the setup (part and tester) - vibration, bumping, jarring, etc., during the test.

6.0.4 Evaluation of the hardness of production parts shall be based on the average of three valid readings.

6.0.4.1 If the average of three readings is less than one hardness point outside the acceptable range for the material being tested, the machine correction factor *may be taken into account when determining the hardness of production parts.

NOTE: Machine correction factor: The difference between the mean of three readings on a standardized hardness test block and the stated hardness of that test block.

* ADDED

6.0.5 When testing curved parts, the following corrections will be added to the Rockwell hardness number:

6.0.5.1 Hardness-curvature relationship for Rockwell normal. Ref. Table IV.

TABLE IV
(ATSM E18-67 TABLES 6 AND 7)

DIAL READINGS	DIAMETERS OF CYLINDRICAL SPECIMENS (a)								
	1/4 IN.	3/8 IN.	1/2 IN.	5/8 IN.	3/4 IN.	7/8 IN.	1 IN.	1-1/4 IN.	1-1/2 IN.
CORRECTIONS TO BE ADDED TO SCALE A, C, & D READINGS (BLACK SCALE)(b)									
20	6.0	4.5	3.5	2.5	2.0	1.5	1.5	1.0	1.0
25	5.5	4.0	3.0	2.5	2.0	1.5	1.0	1.0	1.0
30	5.0	3.5	2.5	2.0	1.5	1.5	1.0	1.0	0.5
35	4.0	3.0	2.0	1.5	1.5	1.0	1.0	0.5	0.5
40	3.5	2.5	2.0	1.5	1.0	1.0	1.0	0.5	0.5
45	3.0	2.0	1.5	1.0	1.0	1.0	0.5	0.5	0.5
50	2.5	2.0	1.5	1.0	1.0	0.5	0.5	0.5	0.5
55	2.0	1.5	1.0	1.0	0.5	0.5	0.5	0.5	0
60	1.5	1.0	1.0	0.5	0.5	0.5	0.5	0	0
65	1.5	1.0	1.0	0.5	0.5	0.5	0.5	0	0
70	1.0	1.0	0.5	0.5	0.5	0.5	0.5	0	0
75	1.0	0.5	0.5	0.5	0.5	0.5	0	0	0
80	0.5	0.5	0.5	0.5	0.5	0	0	0	0
85	0.5	0.5	0.5	0	0	0	0	0	0
90	0.5	0	0	0	0	0	0	0	0
CORRECTIONS TO BE ADDED TO SCALE B & E READINGS (RED SCALE)(b)									
0	12.5	8.5	6.5	5.5	4.5	3.5	3.0		
10	12.0	8.0	6.0	5.0	4.0	3.5	3.0		
20	11.0	7.5	5.5	4.5	4.0	3.5	3.0		
30	10.0	6.5	5.0	4.5	3.5	3.0	2.5		
40	9.0	6.0	4.5	4.0	3.0	2.5	2.5		
50	8.0	5.5	4.0	3.5	3.0	2.5	2.0		
60	7.0	5.0	3.5	3.0	2.5	2.0	2.0		
70	6.0	4.0	3.0	2.5	2.0	2.0	1.5		
80	5.0	3.5	2.5	2.0	1.5	1.5	1.5		
90	4.0	3.0	2.0	1.5	1.5	1.5	0.5		
100	3.0	2.5	1.5	1.5	1.0	1.0	0.5		

Hardness-curvature relationship for Rockwell superficial.

TABLE V
(ASTM E18-67 TABLES 13 and 14)

DIAL READINGS	1/8 IN.	1/4 IN.	3/8 IN.	1/2 IN.	5/8 IN.	3/4 IN.	1 IN.
CORRECTIONS TO BE ADDED TO THE "N" SCALE READINGS (BLACK SCALE)(b)							
20	6.0	3.0	2.0	1.5		1.5	1.5
25	5.5	3.0	2.0	1.5		1.5	1.0
30	5.5	3.0	2.0	1.5		1.5	1.0
35	5.0	2.5	2.0	1.5		1.0	1.0
40	4.5	2.5	1.5	1.5		1.0	1.0
45	4.0	2.0	1.5	1.0		1.0	1.0
50	3.5	2.0	1.5	1.0		1.0	0.5
55	3.5	2.0	1.5	1.0		0.5	0.5
60	3.0	1.5	1.0	1.0		0.5	0.5
65	2.5	1.5	1.0	0.5		0.5	0.5
70	2.0	1.0	1.0	0.5		0.5	0.5
75	1.5	1.0	0.5	0.5		0.5	0
80	1.0	0.5	0.5	0.5		0	0
85	0.5	0.5	0.5	0.5		0	0
90	0	0	0	0		0	0
CORRECTIONS TO BE ADDED TO THE "T" SCALE READINGS (RED SCALE)(b)							
20	13.0	9.0	6.0	4.5	3.5	3.0	2.0
30	11.5	7.5	5.0	4.0	3.5	2.5	2.0
40	10.0	6.5	4.5	3.5	3.0	2.5	2.0
50	8.5	5.5	4.0	3.0	2.5	2.0	1.5
60	6.5	4.5	3.0	2.5	2.0	1.5	1.5
70	5.0	3.5	2.5	2.0	1.5	1.0	1.0
80	3.0	2.0	1.5	1.5	1.0	1.0	0.5
90	1.5	1.0	1.0	0.5	0.5	0.5	0.5

NOTES FOR TABLES IV AND V

(A) When testing cylindrical specimens, the accuracy of the test will be seriously affected by alignment of elevating screws, V-anvil, penetrators, surface finish, and the straightness of the cylinder.

(B) These corrections are approximate only and represent the averages, to the nearest 0.5 Rockwell number, of numerous actual observations.

6.0.5.2 BRINELL

6.0.5.3 The Brinell hardness test method consists of forcing a hardened steel ball into a specimen by using a definite pressure. The diameter of ball, in tests permitted at or for the Fort Worth Operation, is 10 mm and the pressures are 500, 1500 and 3000 Kilograms in magnitude.

Step 1: The specimen is placed on the anvil and elevated to the indenter, or, the portable tester is securely clamped onto the part. All precautions outlined for Rockwell hardness testing are applicable.

Step 2: Pressure is applied as specified in Table II. The load must be allowed to act for from 10 to 15 seconds. The pressure is then removed, the anvil lowered and the part removed.

Step 3: The diameter of the indentation is now measured. For this purpose a special purpose 20 power scope with a graduated reticle is employed. The indentation is measured to the nearest 0.05 mm. Caution must be exercised in reading the brinellscope.

Step 4: The diameter will be converted to a BHN by the use of conversion tables. Reference Tables VI and VII.

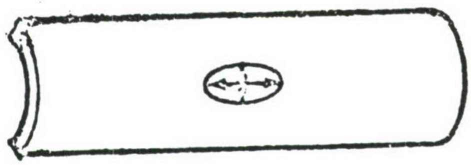


FIGURE II

NOTE: Whenever possible, a flat area shall be ground for testing instead of testing on a curved surface. Ref. Figure II.

6.0.5.4 If the edge of the impression is not well defined, the surface smoothness or the applied load is incorrect.

6.0.5.5 The diameter of the ball shall be 10 mm or .3937 inches ± 0.0002 inches in all directions.

6.0.5.6 The diameter of the impression shall range between 2.50 mm and 6.00 mm. When diameters are out of this range, either, the applied load or the method shall be changed.

6.0.5.7 When indentations are made on a curved surface, the minimum radius of curvature of the surface shall be not less than one inch for the 10 mm dia. ball.

6.0.5.8 When measuring the diameter of the impression on curved surface the two major axes will be measured and averaged and the average dia. converted to a BHN.

TABLE VI

ROCKWELL						BRINELL 10 mm DIA BALL 3000 Kg LOAD		TENSILE STRENGTH	ROCKWELL NORMAL	
NORMAL			SUPERFICIAL							
C	A	D	15N	30N	45N	DIA	BHN	KSI	C	D
68	85.6	76.9	93.2	84.4	75.4	-	-	-	68	-
67	85.0	76.1	92.9	83.6	74.2	-	-	-	67	-
66	84.5	75.4	92.5	82.8	73.3	-	-	-	66	-
65	83.9	74.5	92.2	81.9	72.0	-	-	-	65	-
64	83.4	73.8	91.8	81.1	71.0	-	-	-	64	-
63	82.8	73.0	91.4	80.1	69.9	-	-	-	63	-
62	82.3	72.2	91.1	79.3	68.8	-	-	-	62	-
61	81.8	71.5	90.7	78.4	67.7	-	-	-	61	-
60	81.2	70.7	90.2	77.5	66.6	-	-	-	60	-
59	80.7	69.9	89.8	76.6	65.5	-	-	-	59	-
58	80.1	69.2	89.3	75.7	64.3		615	-	58	-
57	79.6	68.5	88.9	74.8	63.2		595	-	57	-
56	79.0	67.7	88.3	73.9	62.0		577	-	56	-
55	78.5	66.9	87.9	73.0	60.9	2.54	560	301	55	-
54	78.0	66.1	87.4	72.0	59.8	2.64	543	292	54	-
53	77.4	65.4	86.9	71.2	58.6	2.68	525	283	53	-
52	76.8	64.6	86.4	70.2	57.4	2.70	512	273	52	-
51	76.3	63.8	85.9	69.4	56.1	2.75	496	264	51	-
50	75.9	63.1	85.5	68.5	55.0	2.79	481	255	50	-
49	75.2	62.1	85.0	67.6	53.8	2.83	469	246	49	-
48	74.7	61.4	84.5	66.7	52.5	2.87	455	237	48	-
47	74.1	60.8	83.9	65.8	51.4	2.90	443	229	47	-
46	73.6	60.0	83.5	64.8	50.3	2.94	432	222	46	-
45	73.1	59.2	83.0	64.0	49.0	2.98	421	215	45	-
44	72.5	58.5	82.5	63.1	47.8	3.02	409	208	44	-
43	72.0	57.7	82.0	62.2	46.7	3.05	400	201	43	-
42	71.5	56.9	81.5	61.3	45.5	3.09	390	194	42	-
41	70.9	56.2	80.9	60.4	44.3	3.13	381	188	41	-

TABLE VI (Cont'd)

ROCKWELL						BRINELL 10 mm DIA BALL 3000 Kg LOAD		TENSILE STRENGTH	ROCKWELL NORMAL	
NORMAL			SUPERFICIAL							
C	A	D	15N	30N	45N	DIA	BHN	KSI	C	B
40	70.4	55.4	80.4	59.5	43.1	3.15	371	181	40	-
39	69.9	54.6	79.9	58.6	41.9	3.20	362	176	39	-
38	69.4	53.8	79.4	57.7	40.8	3.25	353	170	38	-
37	68.9	53.1	78.8	56.8	39.6	3.29	344	165	37	-
36	68.4	52.8	78.3	55.9	38.4	3.32	336	160	36	-
35	67.9	51.5	77.7	55.0	37.2	3.37	327	155	35	-
34	67.4	50.8	77.2	54.2	36.1	3.42	319	150	34	-
33	66.8	50.0	76.6	53.3	34.9	3.45	311	147	33	-
32	66.3	49.2	76.1	52.1	33.7	3.50	301	142	32	-
31	65.8	48.4	75.6	51.3	32.5	3.55	294	139	31	-
30	65.3	47.7	75.0	50.4	31.3	3.59	286	136	30	-
29	64.3	47.0	74.5	49.5	30.1	3.64	279	132	29	-
28	64.3	46.1	73.9	48.6	28.9	3.69	271	129	28	-
27	63.8	45.2	73.3	47.7	27.8	3.73	264	126	27	-
26	63.3	44.6	72.8	46.8	26.7	3.78	258	123	26	-
25	62.8	43.8	72.2	45.9	25.5	3.82	253	120	25	-
24	62.4	43.1	71.6	45.0	24.3	3.86	247	118	24	-
23	62.0	42.1	71.0	44.0	23.1	3.88	243	115	23	100
22	61.5	41.6	70.5	43.2	22.0	3.93	237	112	22	99
21	61.0	40.9	69.9	42.3	20.7	3.96	231	110	21	98.5
20	60.5	40.1	69.4	41.5	19.6	4.02	226	107	20	97.8
-	-	-	-	-	-	4.07	219	103	-	96.7
-	-	-	-	-	-	4.15	212	100	-	95.5
-	-	-	-	-	-	4.23	203	97	-	93.9
-	-	-	-	-	-	4.33	194	93	-	92.3
-	-	-	-	-	-	4.40	187	90	-	90.7
-	-	-	-	-	-	4.50	179	88	-	89.5
-	-	-	-	-	-	4.59	171	85	-	87.1
-	-	-	-	-	-	4.67	165	83	-	85.5
-	-	-	-	-	-	4.77	158	81	-	83.5
-	-	-	-	-	-	4.85	152	78	-	81.7

Conversion Tables for Aluminum

TABLE VII

ALUMINUM ALLOY (1)	FORM GAGE SPEC.	CONDITION (2)	HARDNESS (Note 3) ROCKWELL E Max.	HARDNESS (Note 3) ROCKWELL E Min.	BRINELL (500 kgm-10 mm) Number Min.	BRINELL (500 kgm-10 mm) Diameter Max.
2014	Sheet & Bar	0	80			
	Plate	T6		103		
	Forging	T6		103	125	2.25
	Extrusions	T4		103	100	2.50
Bare	Sheet &	0	70			
2024	Plate	All T		90		
Clad	Sheet	T3,T4,T6		90		
	(thru	T81,T84,		90		
	0.135)	T86,T851		90		
	Plate			90		
2219	Sheet &	T62,T81, T87		85		
2618	Extrusion	T61			115	2.40
	Forging	T61			115	2.40
6061	Forging	T6			80	2.80
	(QQ-A-367)	0	40			
	All other	T4		63		
		T6		85		
7075	Sheet and	0	70			
	Plate,	T6		105		
	Extrusion,					
	Bar Forg-		See	B-82	135	2.15
	ing					
	(QQ-A-367)	T73	Note 5	B-69		
7178	Sheet	0	70			
		T6		109		
7079	Bar,	T6,T65		B-82	135	2.15
	Extrusion,	T6,T65		B-82	135	2.15
	Forging,	T6,T65		B-82	135	2.15
	Plate	T6,T65		B-82	135	2.15
	(QQ-A-367)					

TABLE VIII

FERROUS MATERIALS

4130

HEAT TREAT RANGE PSI	TEMPERING TEMPERATURE	HARDNESS REQUIRED
90 - 125,000	1100° - 1275°	Rb89 - RC27
125 - 150,000	950° - 1125°	Rc27 - 34
150 - 180,000	860° - 1100°	Rc34 - 40
180 - 200,000	700° - 900°	Rc40 - 43

4335 AND 4340

HEAT TREAT RANGE PSI	TEMPERING TEMPERATURE	REQUIRED HARDNESS
125 - 150,000	1100° - 1250°	Rc27-34
150 - 180,000	950° - 1150°	Rc34-40
180 - 200,000	825° - 1050°	Rc40-43
200 - 220,000	700° - 825°	Rc43-46

D6ac

STRESS LEVEL	180-200,000 PSI	200-220,000 PSI	220-240,000 PSI	260-280,000 PSI
First Temper	550°F Min. 1250° Max.	550°F Min. 1150°F Max.	550°F Min. 1060°F Max.	550°F Min. 700°F Max.
Second Temper	1150°F Min. 1250°F Max.	1050°F Min. 1150°F Max.	1000°F Min. 1060°F Max.	550°F Min. 700°F Max.
Required Hardness	Rc40-43	Rc43-46	Rc46-49	Rc50.5-53

17-7 pH AND 15-7 Mo

Material Condition	17-7 PH Cond. TH 1050	PH 15-7 Mo Cond. TH 1050
Hardness Values Rockwell	C41 - 44 A71 - 73 30N 61-63	C42 - 45 A72 - 73.5 30N 61.5-64

17-4 PH

H 1025	H 900	H 950	H 1075
Heat Cond. A material to 1025-1050°F	Heat Cond. A material to 900°-925°F	Heat Cond. A material to 950°-975°F	Heat Cond. A material to 1075°-1100°F
Hardness required: Rc35-40	Hardness re- Castings: Rc40-44. All other Products Rc40-44.	Hardness required Rc38-42	Hardness required: Rc32-37

NOTES FOR TABLE VII :

(1) For alloys not listed above, representative tensile tests are used to determine acceptability of the materials, since no hardness tests are valid.

(2) The conditions of an alloy may be stated in one, two, three or four digits, i.e., T6, T65, T651 and T6511. The minimum hardness in any case shall be the same as the hardness of the first digit (T6).

(3) Do not convert hardness to tensile strength or other mechanical properties. No correlation is valid.

(4) Clad sheet is approximately 2 points lower than bare.

(5) This value is to be used only to determine that material is not in the "0" condition. It does not in any way indicate whether or not the material has been properly heat treated to the -T73 condition.

7.0 EVALUATION PROCEDURE

7.0.1 Parts which have been hardness tested and do not meet the applicable requirements shall be rejected.

7.0.1.1 Parts which do not conform to hardness requirements of the Engineering drawing may be reheat treated as authorized by Process Control and in accordance with QADI instructions.

7.0.1.2 All parts reprocessed shall be hardness tested in accordance with the requirements of this procedure.

8.0 ACCEPT-REJECT CRITERIA

8.0.1 Parts and raw stocks which conform to the applicable hardness requirements shall be accepted. Rejected parts shall be processed in accordance with para. 7.0.1.1.

9.0 POST INSPECTION REQUIREMENTS

9.0.1 Quality Control personnel shall be responsible for maintaining records of calibration of all equipment used in performing hardness test in accordance with this procedure. Recc [illegible]s of accuracy check are to be retained at equipmen[illegible] [illegible]at[illegible]

9.0.2 Upon completion of hardness testing and evaluation, quality control shall indicate acceptance or rejection on the planning operation sheets or other documents which constitute a part of historical records and shall be retained.

GENERAL DYNAMICS
Fort Worth Division

-NDTS-

NONDESTRUCTIVE TEST STANDARD

NUMBER 30.00-7	ISSUE 2
SUPERSEDING 30.00-7	ISSUE 1
DATE 10 FED 1971	PAGE 1 OF 6

PREPARED [signature]
PROCESS CONTROL [signature]
ENG'R E.K. [signature]
INSP. M. [signature]
AIR FORCE COORDINATED J. W. Johnson

X-RAY INSPECTION OF F-111 WING

This instruction shall be filed with and become an extension of NDTS 30.00.

1. The F-111 lower wing splice area and inboard pivot pylon lower sub-structure shall be inspected per this NDTS before the upper wing skins are installed.

2. Figure I shows typical set-up for inspecting the sub-structure.

3. Figure II shows the view numbering system to be used in the inspection of the lower inboard pivot pylon sub-structure. Table I contains exposure data for these views.

4. Figure III shows typical set-up for inspecting the wing splice area.

5. Table II contains exposure data for wing splice area views.

6. Figure IV shows the view numbering system to be used in inspecting the wing splice area.

7. Any evidence of cracks or other abnormalties caused by manufacturing processes shall be cause for rejection.

8. The exposure data contained in this NDTS gave optimum results with the equipment used in development (Sperry 275 KV, 10 MA unit). Other sources may require adjustment in exposures. The exposure must produce a film density of 1.5 to 3.5 H&D in the area of interest.

9. Inspect the splice area and inboard pylon areas for cracks. A special GD/QC-275-100 penetrameter is to be used in splice area, and regular "AL" penetrameter with shim will be used on pylon shots.

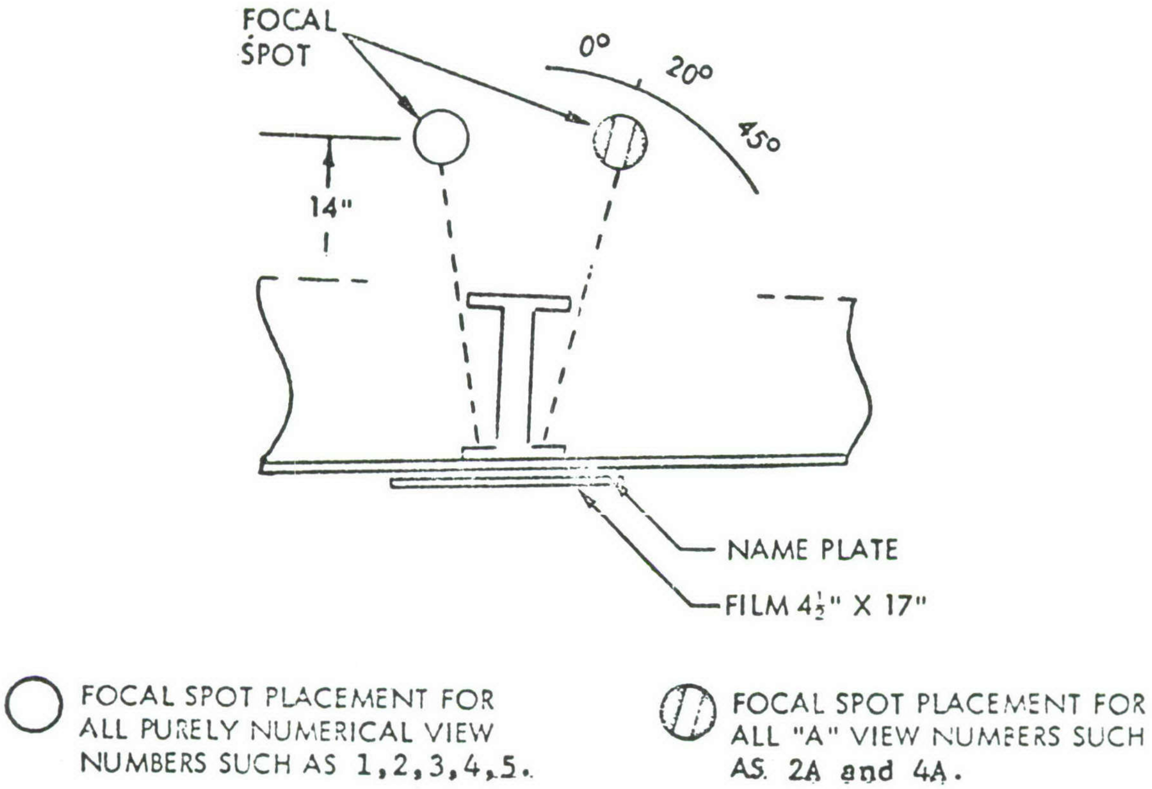
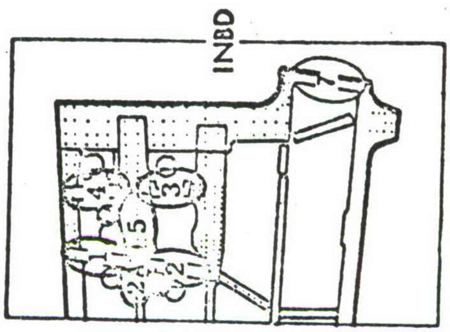


FIGURE I

1. All pylon shots made without upper skin in place, will require an "AL" penetrameter and applicable penetrameter block.

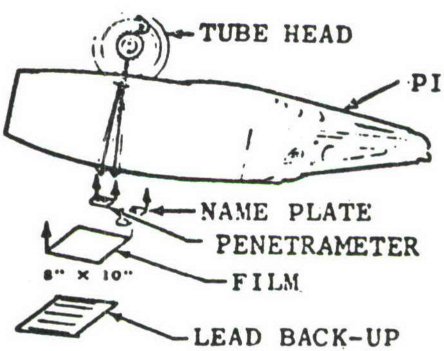
2. Both regular and "A" shots are to be made approximately 20° off vertical.



(NO) = VIEW NUMBER

O = FOCAL SPOT PLACEMENT, REGULAR SHOT
(⊘) = FOCAL SPOT PLACEMENT, "A" SHOT.

FIGURE II
LEFT WING ILLUSTRATED. SAME VIEWS REQUIRED ON RIGHT WING.



HEAD PLACEMENT DIRECTLY ABOVE X-RAY FILM.

FIGURE III
SET-UP FOR X-RAY INSPECTION OF F-111
WING SPLICE AREA - TYPICAL FOR ALL
SHOTS

TABLE I

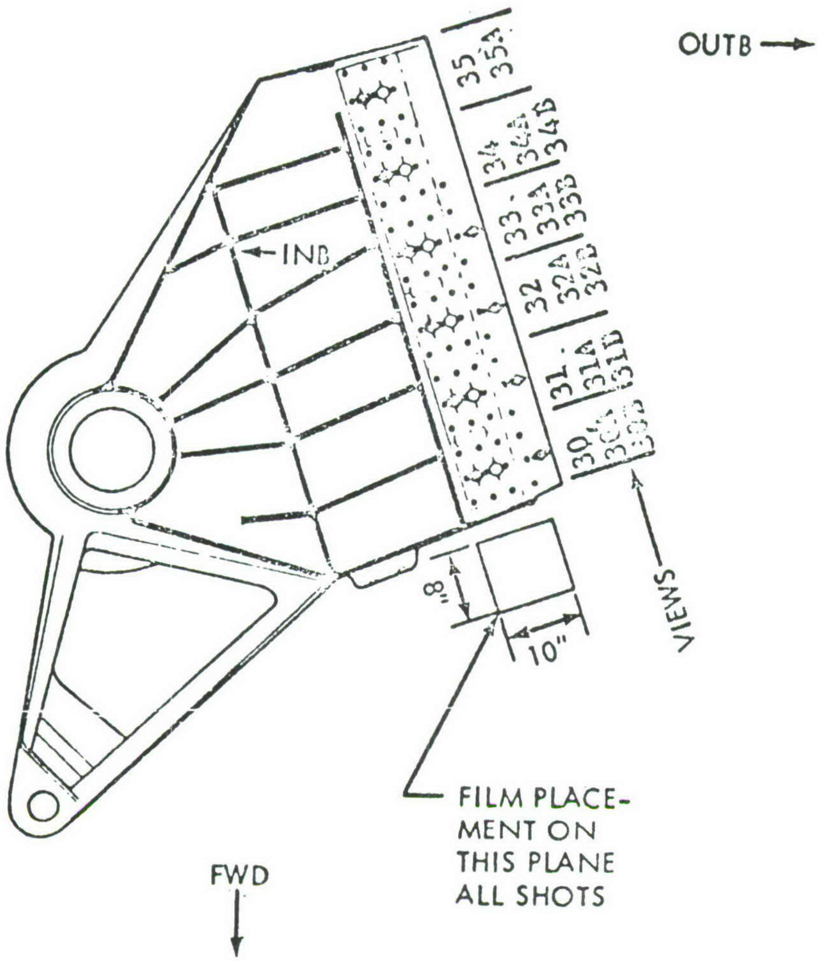
BOTTOM VIEWS INBD. PYLON

VIEW	APPROX. KV	MA	TIME	FOCAL SPOT DIST TO NEAR SURFACE	BEAM ANGLE	FILM KODAK	GENERAL NOTES
1	155	5	1 Min.	14"	20° outbd	'M' Type	1. Name plate must reflect A/C No. specify lower surface - right or left hand part - type of structure being inspected: pylon shots and view No.
2	155	5	1 Min.	14"	20° outbd	'M' Type	
2A	185	5	1 Min.	14"	20° aft	'M' Type	
3	160	5	2 Min.	14"	20° inbd	'M' Type	
4	155	5	1 Min.	14"	20° inbd	'M' Type	
4A	155	5	1 Min.	14"	20° outbd	'M' Type	
5	180	5	1 Min.	14"	45° fwd	'M' Type	

VIEW	KV	MA	TIME	FILM	DISTANCE (INCHES)
30	210	10	2 Min.	Kodak M	6
30A	210	10	2 Min.	Kodak M	6
30B	210	10	2 Min.	Kodak M	6
31	210	10	2 Min.	Kodak M	6
31A	210	10	2 Min.	Kodak M	6
31B	210	10	2 Min.	Kodak M	6
32	210	10	2 Min.	Kodak M	6
32A	210	10	2 Min.	Kodak M	6
32B	210	10	2 Min.	Kodak M	6
33	210	10	2 Min.	Kodak M	6
33A	210	10	2 Min.	Kodak M	6
33B	210	10	2 Min.	Kodak M	6
34	190	10	2 Min.	Kodak M	6
34A	190	10	2 Min.	Kodak M	6
34B	190	10	2 Min.	Kodak M	6
35	190	10	2 Min.	Kodak M	6
35A	190	10	2 Min.	Kodak M	6

TABLE II

EXPOSURE DATA FOR X-RAY INSPECTION OF F-111
WING SPLICE AREA



= Head position for Views 30, 31, 32, 33, 34, and 35.

= Head position for Views 30A, 31A, 32A, 33A, 34A, and 35A.

= Head position for Views 30B, 31B, 32B, 33B, and 34B.

= Typical nameplate placement. Shown for Views 30 thru 35. Move outboard (progressively) for A and B shots.

FIGURE IV

VIEW IDENTIFICATION - WING PIVOT SPLICE

APPENDIX D

ULTRASONIC INSPECTION, METHOD OF

1.0 SCOPE

1.0.1 This standard establishes specific process requirements, inspection procedures or techniques, and quality standards for ultrasonic inspection of wrought metals.

1.1 APPLICATION

1.1.1 The inspection requirements referenced herein are applicable to parts or materials when conformance to the requirements of this standard is required by engineering drawing, procurement specification, purchase order (PO), contract, Nondestructive Test Standard (NDTS), etc., and as required in the manufacturing process to maintain quality standards.

1.1.2 The required ultrasonic classification, as defined herein, shall be shown on the engineering drawing, material specification or PO, Outside Production Operation Sheet (OPOS), contract, etc., for parts or materials requiring inspection to the requirements of this standard.

1.1.3 When necessary for certain materials or parts, and/or when ultrasonic inspection is required, but is not shown on the engineering drawing, material specification, PO, OPOS, contract, etc., General Dynamics, Convair Aerospace Division, Fort Worth Operations (GD/CA/FW) Process Control shall issue specific inspection requirements and procedures (NDTS's) to perform the inspection. When prior ultrasonic inspection has been performed, a certified statement of inspection method and quality level may be submitted to CA/FW Process Control for approval. All such materials or parts inspected in conformance with the requirements of paragraph 1.1.3 shall be released only after authorization has been issued by CA/FW Process Control.

1.1.4 Ultrasonic inspection to the requirements of this standard shall be accomplished on all material by the vendor (supplier) or CA/FW, as specified in conformance with contractual requirements.

1.2 MATERIAL AND/OR PART DESCRIPTION

1.2.1 The inspection requirements referenced herein are applicable to wrought metals, and parts made therefrom, and as specified in accordance with paragraph 1.1.1.

1.3 AREA TO BE INSPECTED

1.3.1 Unless otherwise specified on the engineering drawing or applicable procurement specifications, inspection requirements shall be as specified herein:

1.3.1.1 Material, at each level of production, for steel, titanium, aluminum, and magnesium alloy parts requiring Class AA or Class A ultrasonic quality shall be 100% ultrasonically inspected as specified by engineering specification, purchase order, or NDTS.

1.3.1.2 Steel, titanium, aluminum, and magnesium alloy material used in parts requiring Class B quality shall be inspected in accordance with the inspection requirements established by PO, OPOS, CA/FW Process Control, engineering specification, or specific NDTS.

1.3.2 Flat Stock (rolled or forged plate)

1.3.2.1 All flat stock (rolled or forged plate) 1/2 inch or greater in thickness shall be 100% ultrasonically inspected.

1.3.2.2 All flat stock under 1/2 inch in thickness requiring ultrasonic inspection shall be covered by a technique data sheet provided by the supplier and approved by CA/FW Process Control.

1.3.3 Rectangular Bar Stock

1.3.3.1 All rectangular bar stock having a minimum dimension of 1/2 inch or greater shall be 100% ultrasonically inspected by method(s) provided by the supplier and approved by CA/FW Process Control.

1.3.3.2 All rectangular bar stock having a minimum dimension less than 1/2 inch requiring ultrasonic inspection shall be covered by a technique data sheet provided by the supplier and approved by CA/FW Process Control.

1.3.4 Round Bar Stock

1.3.4.1 All round bar stock shall be 100% ultrasonically inspected when required by engineering drawing, PO, OPOS, or CA/FW Process Control.

1.4 DEFECT DESCRIPTION

1.4.1 Any discontinuity caused by manufacturing process shall be subject to further evaluation regardless of the size, shape or orientation.

1.5 TECHNIQUE DESCRIPTION AND TEST PROCEDURES

1.5.1 Method

1.5.1.1 Unless otherwise approved by CA/FW Process Control, the inspection of raw materials requiring ultrasonic inspection to the requirements of this standard shall be performed using the immersion method.

1.5.1.2 On complex or contoured shapes, contact inspection may be utilized. Use of contact inspection must be approved, however, by CA/FW Process Control.

1.5.2 Water Travel Distance

1.5.2.1 Water travel distance shall be three inches, $\pm$ 1/16 inch, unless otherwise specified herein or otherwise approved by CA/FW Process Control.

1.5.2.2 For extremely long metal travel distances, the water travel distance shall be such that the second front reflection will not appear between the initial front and back reflections.

1.5.2.3 For both standardization and part inspection, the water travel shall be the same.

1.5.3 Test Frequencies, Sensitivities and Crystals

1.5.3.1 The ultrasonic frequencies, sensitivities, and crystals which present the most accurate definition of the required material quality shall be employed and, as applicable, noted on the technique data sheet.

1.5.4 Linearity Characteristics

1.5.4.1 Determination of the linearity characteristics of the ultrasonic unit shall be accomplished at the operating frequency and sensitivity (gain) level employed in the inspection process.

1.5.4.2 The method used in the determination of the linearity characteristics of the ultrasonic unit (such as Lyn-o-check method, ASTM reference ball and reference block method) shall be noted on the technique data sheet.

1.5.5 Discontinuities, Evaluation of

1.5.5.1 The necessary reference standards are described in paragraphs 3.3.1.1 thru 3.3.1.10 and multiple scans from at least two surfaces shall be used as needed to accurately define discontinuities.

1.5.5.2 The inspection techniques used to evaluate discontinuities located by scanning shall be capable of determining their size, extent, and conformance to the required Ultrasonic Classification Standard.

1.5.6 Scanning

1.5.6.1 Generally, initial scanning shall be performed perpendicular to the inspection surface and shall be so oriented as to completely evaluate the entire material.

1.5.6.2 For round or cylindrical parts, a minimum of two scans shall be employed in such a manner as to detect all discontinuities that lie parallel to, or at an angle to, the longitudinal axis of the part.

1.5.6.3 The initial scan on die forgings shall be conducted perpendicular to the parting plane. Detailed procedures on complex die forgings shall be supplied for approval by CA/FW.

2.0 SPECIFICATION CONFORMANCE

2.0.1 The procedures defined in this NDTS require compliance with the contractual requirements of the following specifications:

2.0.1.1 FPS-0018, Quality Requirements for Ultrasonic Inspection, Process Specification for

2.0.1.2 FPS-1041, Engineering, Processing and Inspection Requirements for Structural Forgings

2.0.1.3 MIL-C-45662, Calibration System Requirements

2.0.1.4 FMS-1010, Aluminum Alloy 2024 Plate, Special Quality

2.0.1.5 FMS-1075, Aluminum Alloy 2024 Plate, 2.0 Through 3.0 Inch Thick, Special Short Transverse Quality

2.0.1.6 FMS-1079, Aluminum Alloy 2024 Plate (2.000 - 6 inches Thick) Improved Quality

2.0.1.7 FMS-1011, Procurement Specification for Steel, Cr-Mo-V-Ni, Type D6ac

2.0.1.8 FMS-1012, Procurement Specification for Steel, 4330 Vanadium Modified, Vacuum Melted

2.0.1.9 FMS-1059, Procurement Specification for Titanium Alloy, Ti-6Al-6V-2Sn, Bar and Forgings

2.0.1.10 FMS-1060, Procurement Specification for Titanium Alloy, Ti-6Al-6V-2Sn, Sheet, Strip and Plate

2.0.1.11 ASTM E-127-64, Recommended Practice for Fabricating and Checking Aluminum Alloy Ultrasonic Standard Reference Blocks

2.0.1.12 QADI Q-101, Certification of Nondestructive Testing Personnel (CA/FW personnel only)

2.0.2 Technique Data Sheet

2.0.2.1 A technique data sheet (See Figure 1) shall be completed and submitted for approval to CA/FW Process Control by the vendor (supplier) or CA/FW applicable to all material and/or parts requiring ultrasonic inspection to the requirements of this standard. Specific NDTS(s) shall be provided as required by CA/FW Process Control.

2.0.2.2 All equivalent technique data sheets, or revisions and/or modifications to the required technique data sheets, as proposed by the vendor (supplier) or CA/FW, shall be submitted to CA/FW Process Control for approval.

2.0.2.3 After CA/FW Process Control approval of the equivalent technique data sheet(s) or revisions and/or modifications to the required technique data sheet, the requirements as prescribed in paragraph 2.0.2.1 shall be complied with.

2.0.2.4 Each technique data sheet shall be identified with a control number issued by CA/FW Process Control.

2.1 CLASSIFICATION

2.1.1 Ultrasonic quality levels are defined into Classes AA, A, B and C and are defined as follows:

2.1.2 Class AA Areas

2.1.2.1 Discontinuity indications in excess of the response from a 3/64 inch diameter flat bottom hole, at the estimated discontinuity depth shall not be acceptable.

2.1.2.2 Discontinuity indications greater than the response from a #1 test block or 10% of a 3/64 inch diameter flat bottom hole at the discontinuity depth shall not have their centers closer than one inch or exhibit a dimension greater than 1/8 inch.

2.1.2.3 Hash or sonic noise shall not exceed the response height received from a #1 test block or 10% of a 3/64 inch diameter flat bottom hole at the estimated discontinuity depth.

7.1.2.4 With the instrument set so that the first back reflection from the correct test block is at 80% of the screen saturation adjusted for non-linearity, the material will be inspected for loss of back reflection. The back reflection pattern of acceptable material shall remain at 50% or more of full screen saturation.

7.1.3 Class A Areas

7.1.3.1 Discontinuity indications in excess of the response from a 5/64 inch diameter flat bottom hole at the estimated discontinuity depth shall not be acceptable.

7.1.3.2 Multiple indications in excess of the response from a 3/64 inch diameter flat bottom hole shall not have their indicated centers closer than one inch.

7.1.3.3 Elongated (stringer) type defects in excess of one inch in length shall not be acceptable if at any point along the length the discontinuity indication is equal to or greater than the response from a 2/64 inch diameter flat bottom hole for stainless and alloy steels, and a 3/64 inch diameter flat bottom hole for other metals.

7.1.3.4 Multiple discontinuities giving an indication less than the response from a 3/64 inch diameter flat bottom hole are acceptable only if the back reflection pattern is 50% or more of the back reflection pattern of sound material of the same geometry. The sound beam must be normal to the front and back surfaces to insure that loss of back reflection is not caused by surface roughness, surface waviness, or part geometry variation.

7.1.4 Class B Areas

7.1.4.1 Discontinuity indications in excess of the response from a 8/64 inch diameter flat bottom hole at the estimated discontinuity depth shall not be acceptable.

7.1.4.2 Discontinuity indications in excess of the response from a 5/64 inch diameter flat bottom hole at the estimated discontinuity depth shall not have their indicated centers closer than one inch.

2.1.4.3 Elongated (stringer) type defects in excess of one inch in length shall not be acceptable if at any point along the length the discontinuity indication is equal to or greater than the response from a 5/64 inch diameter flat bottom hole.

2.1.4.4 Multiple discontinuities giving an indication less than the response from a 5/64 inch diameter flat bottom hole are acceptable only if the back reflection pattern is 50% or more of the back reflection pattern of sound material of the same geometry. The sound beam must be normal to the front and back surfaces to insure that the loss of back reflection is not caused by surface roughness, surface waviness, or part geometry variation.

2.1.5 Class C Areas

2.1.5.1 Discontinuity indications in excess of the response from an 8/64 inch diameter flat bottom hole at the estimated discontinuity depth shall not be acceptable.

3.0 <u>CAPABILITIES, EQUIPMENT AND ACCESSORIES</u>

3.0.1 The following capabilities, equipment, and accessories are required to perform ultrasonic inspection in accordance with the requirements of this standard.

3.1 Facility Approval

3.1.1 All vendor laboratories, "in process" inspection techniques, and nondestructive test personnel must be approved by CA/FW prior to furnishing any nondestructive test services controlled by this standard.

3.1.2 Each ultrasonic test facility shall have the capability of meeting the applicable requirements of this standard and must be approved by CA/FW and periodically audited by CA/FW Quality Assurance.

3.1.3 The ultrasonic facilities shall be capable of performing the inspection required by this standard.

3.2 Equipment

3.2.1 The equipment shall include all electronic instrumentation mechanical devices, and calibration accessories required to meet the requirements of this standard.

3.2.2 The equipment shall be capable of producing, receiving, amplifying, displaying, and gating electrical pulses at the required frequencies and energy levels.

3.2.3 The electronic equipment shall be capable of producing frequencies and sensitivities which are adequate for the required quality level classification of Section 2.1.

3.2.4 A voltage regulator shall be employed on the power source to prevent variation of line voltage.

3.2.5 For immersed scanning, an immersion tank and scanning device (manipulating equipment and traversing equipment) shall be required.

3.2.5.1 The manipulating equipment shall be capable of directing the ultrasonic beam in planes necessary to provide maximum discontinuity indication.

3.2.5.2 The traversing equipment shall be rigid and free of backlash to assure that the transducer does not deflect during scanning.

3.3 Reference Standards

3.3.1 The following requirements shall be complied with unless otherwise authorized by CA/FW Process Control:

3.3.1.1 Discontinuities shall be compared to calibrated ultrasonic reference blocks.

3.3.1.2 Only those reference blocks calibrated in accordance with a procedure approved or established by CA/FW Process Control shall be used.

3.3.1.3 Reference blocks of the same basic composition of the material under inspection shall be used.

3.3.1.4 The reference blocks shall contain flat bottom holes of standard diameters at the same depths of tne indicated discontinuities within ± 1/16 inch up to 1/4 inch depth, within ± 1/8 inch over 1/4 up to 1 inch depth, within ± 1/4 inch over 1 up to 3 inch depth and within ± 1/2 inch over 3 inch depth.

3.3.1.5 Reference graphs may be used in lieu of reference blocks after specific approval of CA/FW Process Control.

3.3.1.6 Aluminum alloy reference blocks shall conform to the dimensional requirements of ASTM E-127-64.

3.3.1.7 Steel and titanium reference blocks shall conform to the same dimensional requirements as prescribed for aluminum alloy reference blocks.

3.3.1.8 Reference blocks made from 4340 or 4330 V Mod. or D6ac steel alloys in the normalized and tempered condition must be used to inspect any air melted low or medium alloy steel material.

3.3.1.9 Reference blocks made from vacuum melted material in the normalized and tempered condition shall be used to inspect vacuum melted material of the same basic chemical composition.

3.3.1.10 Reference blocks made from commercially pure titanium may be used to inspect titanium alloys.

4.0 <u>PREINSPECTION PART PREPARATION</u>

4.0.1 Material surface conditions shall be as necessary to reliably perform the applicable ultrasonic inspection in accordance with Section 2.1.

4.0.2 The material to be inspected shall be free of loose scale, heavy oxides, grease, oil and foreign matter which could lead to erroneous interpretation of test results.

5.0 <u>CALIBRATION PROCEDURES</u>

5.0.1 All measuring and test equipment used in the inspection of material and/or parts requiring ultrasonic inspection to the requirements of this standard shall be calibrated to control the accuracy of the system performance to assure the applicable quality level.

5.1 STANDARDIZATION OF EQUIPMENT

5.1.1 The signal response from the ultrasonic equipment shall be standardized at 80% of screen saturation with the necessary adjustment to correct for instrument non-linearity.

5.1.2 Standardization shall be on the 3/64 inch diameter flat bottom hole for Class AA inspection, 5/64 inch diameter flat bottom hole for Class A inspection, or 8/64 inch diameter flat bottom hole for all other classes of inspection, and in accordance with paragraphs 3.3.1.1 thru 3.3.1.10.

6.0 <u>INSPECTION PROCEDURES</u>

6.0.1 During the initial inspection of each part, areas where configuration prevents ultrasonic inspection shall be noted on a technique data sheet and submitted to CA/FW Process Control for approval. In some cases specific NDTS(s) shall be provided.

6.0.2 Gate Alarm Level

6.0.2.1 The gate alarm level shall be set to alarm at an amplitude of 10% of the screen saturation for Classes AA and A inspection and 30% of the screen saturation for all other classes of inspection.

6.1 Flat Stock (Rolled or forged plate)

6.1.1 Direction of Sound Propogation

6.1.1.1 All flat stock shall be inspected with the direction of sound propogation parallel to the short transverse direction.

6.1.2 Transducer Travel

6.1.2.1 Transducer travel shall be parallel to the long transverse direction and normal to the stock, unless otherwise specified.

6.1.3 Speed of Travel

6.1.3.1 Speed of travel shall be such that the smallest flaw applicable to the class of ultrasonic inspection will be detected. (Not to exceed 1 surface ft/sec).

6.1.3.2 Specific details of speed of travel shall be noted on the technique data sheet.

6.1.4 Transducer Indexing

6.1.4.1 Transducer indexing shall be parallel to the longitudinal direction of the stock and shall not exceed 80% of the minimum Effective Beam Diameter (EBD) at any depth in the material for each particular transducer and setup.

6.1.5 Normality

6.1.5.1 Transducer normality shall be checked on each immersion tank load of flat stock by the following method:

6.1.5.1.1 Normality shall be checked in regard to the front (entrant) surface of the material under inspection.

6.1.5.1.2 Maintain a water distance of three inches, ± 1/16 inch, or as required in accordance with paragraph 1.5.2.2.

6.1.5.1.3 Angulate the search tube until the signal displayed on the Cathode Ray Tube from the front surface is maximized, and lock the search tube at this position.

6.1.5.1.4 When normalizing, the reject control, db control, or a step function of sensitivity shall be employed to maintain the signal level from the front surface to a point under screen saturation.

6.1.5.1.5 The sweep length control(s) shall be adjusted as necessary to position the Cathode Ray Tube presentation such that the signal from the front surface and the signal from the back surface are displayed clearly on the screen.

6.1.5.1.6 After normalizing, assure that the instrument sensitivity (gain) setting as previously accomplished is maintained as applicable to the required ultrasonic classification.

6.1.6 Back Reflection, Loss of

6.1.6.1 All flat stock requiring Class AA inspection shall be inspected for loss of back reflection by the following procedure unless otherwise specified in an applicable NDTS.

6.1.6.2 This operation shall preceed the evaluation for flaw content.

6.1.6.3 Class AA Inspection

6.1.6.3.1 The back reflection signal from a reference block, its total length being approximately the same as the thickness of the material under inspection, shall be set at 80% of the screen saturation after the proper adjustments for the instrument non-linearity.

6.1.6.3.2 Maintain the same water distance over the stock to be inspected as over the setup block.

6.1.6.3.3 Scan a minimum of three longitudinal passes to the requirements of paragraph 2.1.2.4 of the entire length of each piece of stock as follows:

6.1.6.3.3.1 One down the center and one down each mid-radius line.

6.2 Rectangular Bar Stock

6.2.1 All rectangular bar stock shall be inspected using longitudinal waves.

6.2.2 Dual inspection shall be performed from two adjacent surfaces.

6.2.3 Transducer Travel

6.2.3.1 Transducer travel shall be parallel to the length of the stock.

6.2.4 Speed of Travel

6.2.4.1 Speed of travel shall be such that the smallest flaw applicable to the class of ultrasonic inspection will be detected. (Not to exceed 1 surface ft/sec).

6.2.4.2 Speed of travel shall be noted on the technique data sheet.

6.2.5 Transducer Indexing

6.2.5.1 Transducer indexing shall be perpendicular to the stock length and shall not exceed 80% of the minimum Effective Beam Diameter (EBD) at any depth in the material for each particular transducer and setup. See paragraph 6.1.4.1.

6.2.6 Normality

6.2.6.1 Transducer normality shall be checked on each separate piece of material by the method as prescribed in paragraphs 6.1.5.1.1 thru 6.1.5.1.6.

6.2.7 Back Reflection, Loss of

6.2.7.1 All rectangular bar stock requiring Class AA inspection shall be inspected for loss of back reflection by the procedure as prescribed in paragraphs 6.1.6.2 thru 6.1.6.3.3.1 unless otherwise specified in an applicable NDTS.

6.3 Round Bar Stock

6.3.1 All round bar stock shall be inspected using both longitudinal and shear waves.

6.3.2 Relationship of transducers to bar stock center line.

6.3.2.1 The first inspection pass shall use longitudinal waves.

6.3.2.1.1 The longitudinal waves shall be emitted perpendicular to the length of the stock and passing through its center.

6.3.2.1.2 The distance the physical center of the transducer is off the center of the stock under inspection shall be the Y-distance.

6.3.2.2 The second inspection pass shall use shear waves.

6.3.2.2.1 The shear waves shall be generated within the stock by off setting the physical center of the transducer by the X-distance.

6.3.2.2.2 The X-distance being measured from the center line of the stock.

6.3.2.3 When required by the tables, a third inspection pass shall be made.

6.3.2.3.1 The position of this pass shall be governed by the X'-distance.

6.3.2.3.2 The distance shall be measured from the center line of the stock.

6.3.2.4 Transducer travel shall be parallel to the length of the stock.

6.3.2.5 The stock shall be rotated at such a rate that the smallest flaw applicable to the class of ultrasonic inspection will be detected.(Not to exceed 1 surface ft/sec).

6.3.2.5.1 The rate of rotation shall be noted on the technique data sheet.

6.3.2.6 The stock shall be rotated at such a rate that the transducer shall not travel more than one third of its minimum effective beam diameter per revolution.

6.3.2.7 The parameters as listed in Table I shall be used during the inspection of all round bar stock, unless an alternative procedure has been submitted by the vendor (supplier) or CA/FW and approved by CA/FW Process Control.

TABLE I

BAR DIA.	TRANSDUCER DIAMETER	TRANSDUCER FREQUENCY	MINIMUM WATER DISTANCE	NUMBER OF PASSES	Y DISTANCE	X DISTANCE	X' DISTANCE	Set Up Ref. Block Metal Travel	dB
ALUMINUM ROUND BAR STOCK									
1"	.375"	10 MHz	2"	2	0	.100"	None	.0025	9
1" - 2"	.375"	10 MHz	2"	2	0	.200"	None	.0088	9
2" - 3"	.500"	10 MHz	2"	2	0	.250"	None.	.0125	9
3" - 4"	.500"	5 MHz	2"	2	0	.300"	None	.0175	6
4" - 5"	.750"	5 MHz	3"	2	.125"	.400"	None	.0225	6
5" - 6"	.750"	5 MHz	3"	3	.125"	.500"	.875"	.0225	6
6" - 7"	.750"	5 MHz	3"	3	.250"	.600"	.950"	.0275	5
7" - 8"	.750"	5 MHz	3"	3	.250"	.700"	1.000"	.0325	5
8" - 9"	.750"	5 MHz	3"	3	.300"	.700"	1.100"	.0375	3
9" - 10"	.750"	5 MHz	3"	3	.300"	.800"	1.200"	.0425	3
10" - 11"	.750"	5 MHz	3"	3	.400"	.900"	1.300"	.0425	2
11" - 12"	.750"	5 MHz	3"	3	.400"	1.000"	1.400"	.0525	1
STEEL ROUND BAR STOCK (Vac. Melt - #3 (Class AA) (Air Melt - #5 (Class A)									
1"	.375"	10 MHz	2"	2	0	.100"	None	.0025	9
1" - 2"	.375"	10 MHz	2"	2	0	.200"	None	.0088	9
2" - 3"	.500"	10 MHz	2"	2	0	.300"	None	.0125	9
3" - 4"	.500"	5 MHz	2"	2	0	.350"	None	.0175	6
4" - 5"	.750"	5 MHz	3"	2	.125"	.450"	None	.0225	6
5" - 6"	.750"	5 MHz	3"	3	.125"	.550"	.900"	.0275	6
6" - 7"	.750"	5 MHz	3"	3	.250"	.600"	.975"	.0275	5
7" - 8"	.750"	5 MHz	3"	3	.250"	.700"	1.025"	.0325	5
8" - 9"	.750"	5 MHz	3"	3	.300"	.800"	1.125"	.0375	3
9" - 10"	.750"	5 MHz	3"	3	.300"	.900"	1.225"	.0425	3
10" - 11"	.750"	5 MHz	3"	3	.400"	1.000"	1.325"	.0475	2
11" - 12"	.750"	5 MHz	3"	3	.400"	1.100"	1.425"	.0525	1
STAINLESS STEEL ROUND BAR STOCK									
1"	.375"	10 MHz	2"	2	0	.100"	None	.0025	8
1" - 2"	.375"	10 MHz	2"	2	0	.200"	None	.0088	8
2" - 3"	.500"	10 MHz	2"	2	0	.300"	None	.0125	8
3" - 4"	.500"	5 MHz	2"	2	0	.350"	None	.0175	5
4" - 5"	.750"	5 MHz	3"	2	.125"	.450"	None	.0225	5
5" - 6"	.750"	5 MHz	3"	3	.125"	.550"	.900"	.0275	5
6" - 7"	.750"	5 MHz	3"	3	.250"	.600"	.975"	.0275	4
7" - 8"	.750"	5 MHz	3"	3	.250"	.700"	1.025"	.0325	4
8" - 9"	.750"	5 MHz	3"	3	.300"	.800"	1.125"	.0325	2
9" - 10"	.750"	5 MHz	3"	3	.300"	.900"	1.225"	.0425	2
10" - 11"	.750"	5 MHz	3"	3	.400"	1.000"	1.325"	.0475	1
11" - 12"	.750"	5 MHz	3"	3	.400"	1.100"	1.425"	.0525	1
TITANIUM ROUND BAR STOCK									
1"	.375"	10 MHz	2"	2	0	.100"	None	.0025	9
1" - 2"	.375"	10 MHz	2"	2	0	.200"	None	.0088	9
2" - 3"	.500"	10 MHz	2"	2	0	.250"	None	.0125	9
3" - 4"	.500"	5 MHz	2"	2	0	.300"	None	.0175	6
4" - 5"	.750"	5 MHz	3"	2	.125"	.400"	None	.0225	6
5" - 6"	.750"	5 MHz	3"	3	.125"	.500"	.875"	.0275	6
6" - 7"	.750"	5 MHz	3"	3	.250"	.600"	.950"	.0275	5
7" - 8"	.750"	5 MHz	3"	3	.250"	.700"	1.000"	.0325	5
8" - 9"	.750"	5 MHz	3"	3	.300"	.700"	1.000"	.0375	3
9" - 10"	.750"	5 MHz	3"	3	.300"	.800"	1.200"	.0425	3
10" - 11"	.750"	5 MHz	3"	3	.400"	.900"	1.300"	.0425	2
11" - 12"	.750"	5 MHz	3"	3	.400"	1.000"	1.400"	.0525	1

7.0 EVALUATION PROCEDURE

7.0.1 Defect indications on the CRT screen which exceed the alarm level that is applicable to the required ultrasonic classification shall be evaluated as follows:

7.0.1.1 Flat Stock (Rolled or forged plate) and Rectangular Bar Stock

7.0.1.1.1 A minimum of two reference blocks shall be employed in each setup.

7.0.1.1.2 One reference block shall have a metal travel distance of one-quarter of an inch or less, the other reference block shall have a metal travel distance that approaches the thickness of the material under inspection.

7.0.1.1.3 Using the reference block that gives the minimum indication on the Cathode Ray Tube as a setup reference, set its indication height at 80% of the screen saturation after the necessary adjustment for instrument non-linearity.

7.0.1.1.4 The gate alarm level shall be set to alarm at an amplitude of 10% of the screen saturation for Classes AA and A inspection and 30% of the screen saturation for all other classes of inspection.

7.0.1.1.5 Indexing shall be set to be equal to 80% of the minimum effective beam diameter.

7.0.1.1.6 All discontinuities giving an indication large enough in amplitude to alarm the gate shall be noted as to depth and their positions shall be marked on all material.

7.0.1.1.7 All marked discontinuities shall be evaluated by comparison with the proper reference block.

7.0.1.1.8 The indications that are of sufficient amplitude shall be measured for length and width using the effective beam diameter at that depth.

7.0.1.1.9 All flaws shall be marked as to position, depth and relative size and shall be dispositioned in accordance with respect to their apparent results on the ultimate product.

7.0.1.1.10 The following marking code shall be used as illustrated in Figure 2.

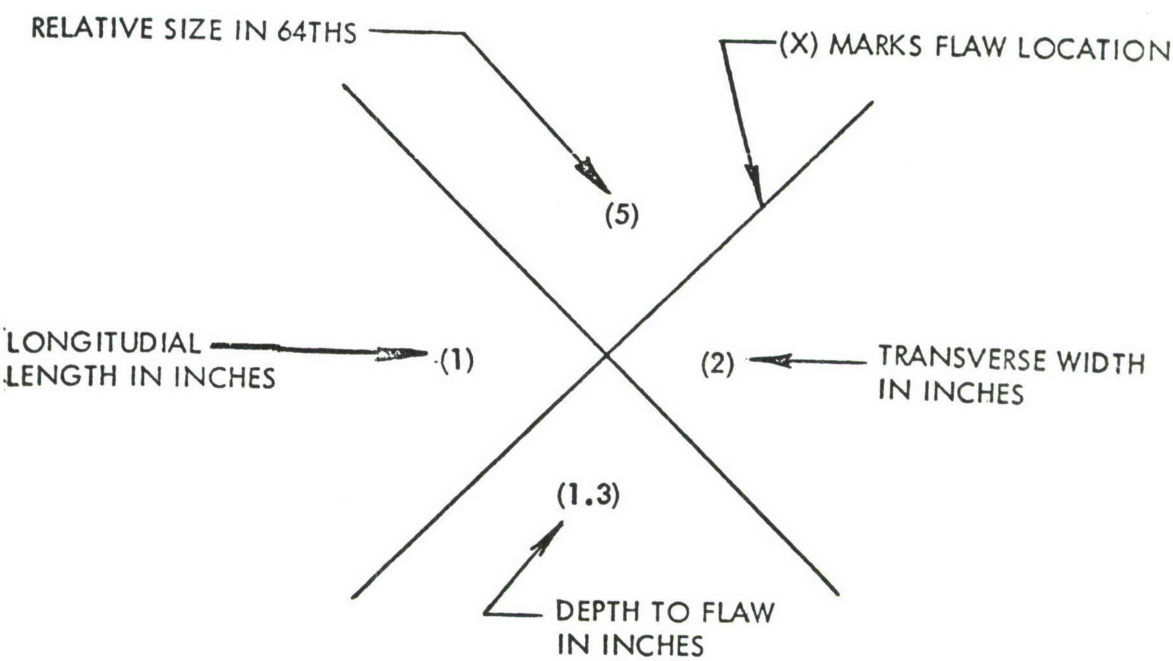


FIGURE 2
EXAMPLE OF MARKING CODE

7.0.1.1.11 All acceptable plate stock shall be identified on both ends.

7.0.1.2 Round Bar Stock

7.0.1.2.1 Marking of flaw indication.

7.0.1.2.1.1 All flaws shall be located as to their longitudinal position along the stock.

7.0.1.2.1.2 Only flaws found by the first inspection pass shall be located as to depth.

8.0 ACCEPT/REJECT CRITERIA

8.0.1 Parts and/or material failing to meet the requirements of this standard shall be rejected.

9.0 POST INSPECTION REQUIREMENTS

9.0.1 Unless otherwise authorized by CA/FW Process Control, each item meeting the applicable ultrasonic requirements of this standard shall be marked with the applicable class designation and this standard number. The marking designation shall be provided by the supplier.

9.0.2 A certified report, signed by the laboratory director or his authorized assistant, shall be furnished in triplicate with each shipment.

9.0.2.1 This report shall include the technique data sheet, the inspection class, this standard number, the inspection method employed, the results, and all other requirements as applicable to the conformance of this standard.

9.0.3 Procurement Quality Assurance personnel shall be responsible for maintaining adequate records of all parts and/or material inspected to the requirements of this standard.

PC Number ______

ULTRASONIC DATA RECORD

COMPANY NAME ______________________ DATE ______________

ADDRESS ______________________ STATE ______________

OPERATOR'S NAME ______________________

PART NO. ____________ PART OR MATERIAL DESCRIPTION ______________

MATERIAL ______________________

SPECIFICATION REQUIREMENTS: CLASSIFICATION ______________

MATERIAL COMPOSITION ______________

EQUIPMENT: MAKE ____________ MODEL ______________

METHOD OF ULTRASONIC APPLICATION

LONGITUDINAL ________ SHEAR ________ OTHER ________

COUPLANT ________ INHIBITOR ______________

IMMERSION ________ CONTACT ________ OTHER ________

WATER TRAVEL ________ INCHES

EQUIPMENT DATA

REFERENCE BLOCKS NUMBER EMPLOYED

	1st SET-UP	2nd SET-UP	3rd SET-UP
BLOCK NO. 1			
BLOCK NO. 2			

BACK REFLECTION SET-UP REFERENCE BLOCK NO. ______________

FREQUENCY OF ULTRASONIC UNIT ______________

	1st	2nd	3rd	
TRANSDUCER FREQUENCY				
COMPOSITION				
SIZE				

MEASURED EFFECTIVE BEAM DIAMETER ____________ INDEX DISTANCE ____________

SCAN RATE (SPEED OF TRAVEL) ________ SURFACE FT/SEC

INSTRUMENT LINEARITY: BEFORE TEST ____________ AFTER TEST ____________

METHOD USED ______________

SPECIAL INSTRUMENTATION OR PROCEDURE(s) - (USE ADDITIONAL SHEETS AS REQUIRED)

____

____

TEST RESULTS: ______________________

____

____

____

____

____

NOTE: Sketches of actual parts and method of application shall be submitted to Process Control as necessary in accordance with paragraph 6.0.1.

FIGURE 1 - TECHNIQUE DATA SHEET

SUPPLEMENT (D)

M186 STANDARD, SERIAL NUMBER FORMAT,

TRACEABILITY

(D)

1.0 APPLICATION

THIS STANDARD IS APPLICABLE ONLY WHEN SPECIFICALLY CALLED OUT ON THE ENGINEERING DRAWING.

2.0 SCOPE

THIS STANDARD ESTABLISHES THE REQUIREMENT AND FORMAT FOR SERIAL NUMBERING OF PARTS FOR TRACEABILITY.

3.0 PURPOSE

THE SERIAL NUMBER SHALL PROVIDE IDENTIFICATION FOR TRACEABILITY TO ESTABLISH A COMPLETE RECORD OF THE PROCESSING OF A PART. RECORDS SHALL BE KEPT IN SUCH A MANNER THAT THE PART SERIAL NUMBER CAN BE RELATED TO ALL THE PARTICULARS OF THE RAW MATERIAL, FORGINGS, ROLLED RINGS, CASTINGS, EXTRUSIONS, PLATES, MACHINING OPERATIONS, MANUFACTURING AND INSPECTION PROCESSES. THE MATERIAL SPECIFICATION AND PART NUMBER IDENTIFICATION SHALL BE RECORDED AGAINST THE SERIAL NUMBER BY THE RAW MATERIAL OR MACHINE PARTS SUPPLIER WHO SHALL FURNISH THE BUYER WITH SUCH INFORMATION UPON REQUEST.

4.0 REQUIREMENTS

4.1 SERIAL NUMBERS SHALL BE LOCATED AS SPECIFIED ON THE ENGINEERING DRAWING. MARKINGS SHALL BE PER FPS-1043, CLASS 1A. HOT IMPRESSED MARKS NOT TO EXCEED 0.060 INCH IN DEPTH OR BY RAISED MARKS FORMED IN THE FORGING DIES MAY BE PERMISSIBLE BY THE ENGINEERING DRAWING. WHEN THE DETAIL PART SERIAL NUMBER IS OBSCURED ON INSTALLATION, A NON-PERMANENT TYPE MARKING SHALL BE PLACED IN A VISIBLE LOCATION AT TIME OF INSTALLATION. THE REQUIREMENT FOR SERIALIZATION DOES NOT AFFECT THE IDENTIFICATION REQUIRED PER MATERIAL SPECIFICATION OR ENGINEERING DRAWING.

4.2 FORMAT

(a) ~~(1) WG 12345 CP~~

~~MACHINE VENDOR IDENTIFICATION LETTERS, THREE LETTERS MAXIMUM (USE FWD OR CF FOR PARTS MACHINED IN HOUSE) TO BE ADDED TO BASIC NUMBER WHEN PART IS MACHINED.~~

~~FORGING VENDORS SERIAL NUMBER OR MATERIAL AS FOLLOWS: PLATE, BAR, ROD, BILLET, INGOT, OR FORGING NUMBER. FIVE DIGIT MAXIMUM.~~

~~SUPPLIERS IDENTIFICATION LETTERS. THREE LETTERS MAXIMUM.~~

(2) LP 0567

EXAMPLE OF SERIAL NUMBERS OR COMBINATION LETTERS AND SERIAL NUMBERS TO BE ASSIGNED BY THE RAW MATERIAL VENDOR OR BY THE MACHINE VENDOR OR CONVAIR AEROSPACE DIVISION MANUFACTURING CONTROL DEPARTMENT.

EXAMPLE OF RAW MATERIAL OR MACHINE PARTS SUPPLIER'S IDENTIFICATION CODE LETTERS. SUPPLIER MAY BE RAW MATERIAL MANUFACTURER, MACHINE VENDOR, OR CONVAIR AEROSPACE DIVISION MANUFACTURING DEPARTMENT.

4.2.1 EXAMPLES OF SERIAL NUMBERS

WG 34576 - "WG", THE FORGING VENDOR, "34576" THE SERIAL NUMBER.

GH 237 - "GH", THE MACHINE VENDOR, "237" THE SERIAL NUMBER.

FW K34567 - "FW", CONVAIR FORT WORTH OPERATION, "K34567" THE SERIAL NUMBER.

THE SUPPLIER'S IDENTIFICATION CODE MAY BE ONE OR MORE LETTERS AND SHALL BE ASSIGNED BY THE PROCUREMENT DEPARTMENT FOR OUTSIDE PRODUCTION PARTS AND BY MANUFACTURING CONTROL DEPARTMENT FOR PARTS MACHINED IN CONVAIR AEROSPACE DIVISION MANUFACTURING PLANT.

4.2.2 THE MACHINED PARTS SUPPLIER SHALL ASSURE THAT THERE ARE NO DUPLICATION OF SERIALIZED PART NUMBERS ASSIGNED TO THE SAME DETAIL PARTS. THE SERIAL NUMBER SHALL BE ASSIGNED IN A SYSTEMATIC MANNER THAT WILL AFFORD READY REFERENCE TO THE COMPLETE ACCOUNT OF THE PROCESSING OF THE PART.

4.2.3 WHEN THE SERIALIZED RAW MATERIAL FOR A PART IS RECEIVED BY THE MACHINE VENDOR OR BY CONVAIR AEROSPACE DIVISION MANUFACTURING PLANT, THE SERIAL NUMBER SHALL BE RECORDED WITH THE MATERIAL SPECIFICATION AND PART NUMBER IDENTIFICATION FOR THEIR RECORDS. THE RAW MATERIAL VENDOR'S SERIAL NUMBER SHALL BE RECORDED AGAINST THE MACHINE VENDOR'S SERIAL NUMBER.

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4.2.4 RAW MATERIAL OF PLATE, BILLET, ROLLED RINGS, BAR, ROD, OR EXTRUSIONS MAY BE RECEIVED BY OUTSIDE PRODUCTION MACHINE VENDORS OR CONVAIR AEROSPACE DIVISION WITHOUT SERIAL NUMBERS. THE RAW MATERIAL SPECIFICATION IDENTIFICATION INCLUDING HEAT NUMBER OR HEAT CODE LETTERS SHALL BE RECORDED AGAINST THE MACHINE VENDOR'S SERIAL NUMBER FOR THEIR RECORDS.

5.0 RECORDS

5.1 RECORDS SHALL BE KEPT IN SUCH A MANNER THAT THE PART SERIAL NUMBER CAN BE TRACED TO THE RAW MATERIAL, MACHINING, MANUFACTURING AND INSPECTION PROCESSES.

5.2 ALL RECORDS COLLECTED IN APPLICATION OF THIS STANDARD BY THE SUPPLIER SHALL REMAIN IN THE FILES OF THE SUPPLIER FOR A MINIMUM PERIOD OF TIME AS REQUIRED BY THE PURCHASE ORDER OR FOR A PERIOD OF NOT LESS THAN THREE YEARS AFTER COMPLETION OF THE PURCHASE ORDER OR THREE YEARS AFTER FINAL SETTLEMENT, IN THE EVENT THE PURCHASE ORDER IS TERMINATED PRIOR TO COMPLETION.

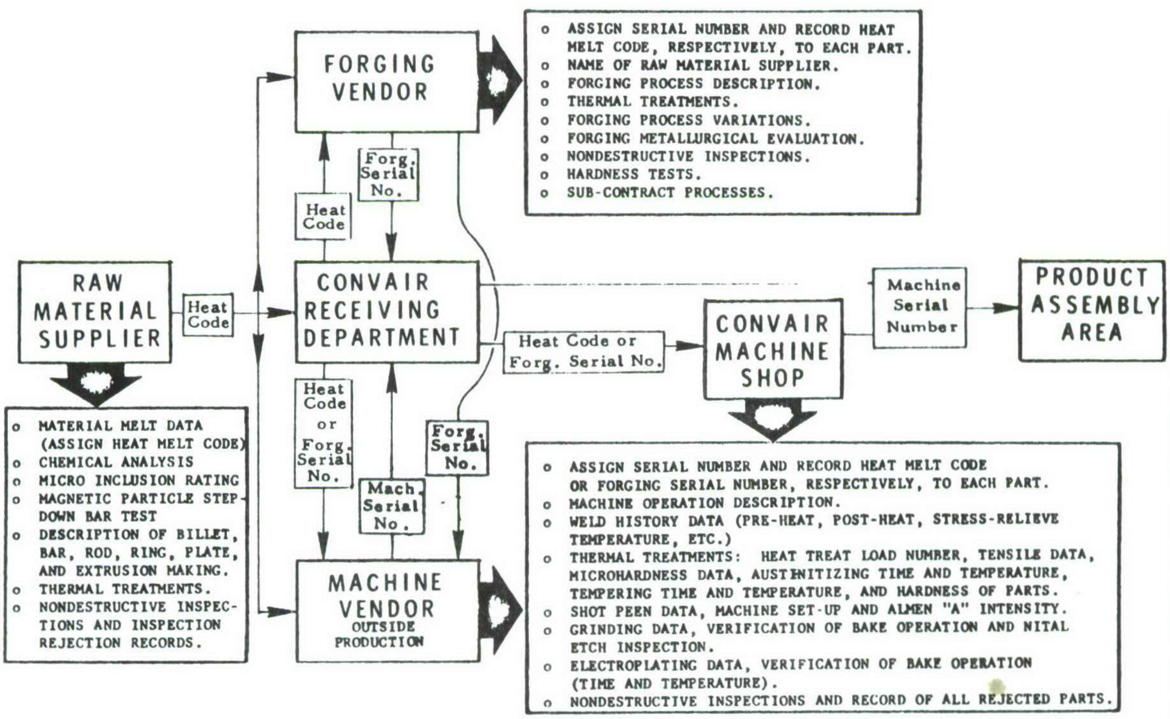
6.0 ENGINEERING REFERENCE

6.1 TYPICAL NOTE SPECIFIED ON ENGINEERING DRAWINGS FOR PERMISSIVE TRACEABILITY SERIALIZATION: "IT IS PERMISSIBLE TO PLACE MANUFACTURER'S SERIAL NO. IN THE LOCATION SHOWN FOR PURPOSES OF TRACEABILITY. SERIAL NUMBER PER M186."

6.2 TYPICAL NOTE SPECIFIED ON ENGINEERING DRAWINGS FOR MANDATORY TRACEABILITY SERIALIZATION: "PART SERIALIZATION REQUIRED FOR (DASH NO). SERIAL NUMBER PER M186."

7.0 FLOW CHART FOR TRACEABILITY

RECORDS DATA TO BE RETAINED BY SUPPLIERS AS MAY BE APPLICABLE PER MATERIALS SPECIFICATION OR ENGINEERING DRAWING



(a) INACTIVE AFTER MAY 1972

ORIGINATING GROUP: DESIGN MATERIALS

APPROVED 25 February 1970 REVISED (D) 10 May 1972

GENERAL DYNAMICS
Fort Worth Division FORT WORTH, TEXAS

STANDARD

SERIAL NUMBER FORMAT, TRACEABILITY

M 186

CONTRACT NO. AF33(657)-13403 | CODE IDENT NO. 81755 | SHEET 2

DEPT 065
FWP2771A-10-65

SUPPLEMENT (E)

PROPOSED REVISION TO MIL-A-8866

DATED 18 AUGUST 1972

USAF DAMAGE TOLERANCE CRITERIA

1.0 Definitions and General Requirements

1.1 Definitions

1.1.1 Degree of Inspectability. The degree of inspectability of each element of safety of flight structure shall be established in accordance with the following definitions.

1.1.1.1 In-Flight Evident Inspectable - Structure is in-flight evident inspectable if the nature and extent of damage occurring in flight will result directly in characteristics which make the flight crew immediately and unmistakably aware that significant damage has occurred and that the mission should not be continued.

1.1.1.2 Ground Evident Inspectable - Structure is ground evident inspectable if the nature and extent of damage being considered will be readily and unmistakably obvious to ground personnel without specifically inspecting the structure for damage.

1.1.1.3 Walkaround Inspectable - Structure is walkaround inspectable if the nature and extent of damage being considered is unlikely to be overlooked by personnel conducting a visual inspection of the structure. This inspection normally shall be a visual look at the exterior of the structure from ground level without removal of access panels or doors and without special inspection aids.

1.1.1.4 Special Visual Inspectable - Structure is special visual inspectable if the nature and extent of damage being considered is unlikely

to be overlooked by personnel conducting a detailed visual inspection of the aircraft for the purpose of finding damaged structure. The procedure may include removal of access panels and doors, and may permit simple visual aids such as mirrors and magnifying glasses. Removal of paint, sealant, etc. and use of NDI techniques such as penetrant, x-ray, etc. are not part of a special visual inspection.

1.1.1.5 Depot or Base Level Inspectable - Structure is depot or base level inspectable if the nature and extent of damage being considered will be detected with a 90% probability at 95% confidence level for slow crack growth structure and with 90% probability at 50% confidence level for fail safe structure. The inspection procedures may include NDI techniques such as penetrant, x-ray, ultrasonic, etc. Accessibility considerations may include removal of those components designed for removal.

1.1.1.6 In-Service Non-Inspectable Structure - Structure is in-service non-inspectable if either damage size or accessibility preclude detection during one or more of the above inspections.

1.1.2 Frequency of Inspection - Frequency of inspection is the number of times that a particular type of inspection is to be conducted during the service life of the aircraft.

1.1.3 Minimum Period of Unrepaired Service Usage - Minimum period of unrepaired service usage is that period of time during which the appropriate level of damage (assumed initial or in-service) is presumed to remain unrepaired and allowed to grow within the structure.

1.1.4 Minimum Required Residual Strength (P_{XX}) - The minimum required residual strength shall be as specified in Paragraph 1.2.2.

1.1.5 Minimum Assumed Initial Damage Size - The minimum assumed initial damage size is the smallest crack-like defect which shall be used as a starting point for analyzing residual strength and crack growth characteristics of the structure.

1.1.6 Minimum Assumed In-Service Damage Size - The minimum assumed in-service damage size is the smallest damage which shall be assumed to exist in the structure after completion of an in-service inspection.

1.1.7 Damage Growth Limit - Damage growth limit is the maximum amount of damage growth allowed within a specified interval so as not to degrade the residual strength below a specified minimum level.

1.1.8 Slow Crack Growth Structure - Slow crack growth structure consists of those design concepts where flaws or defects are not allowed to attain the critical size required for unstable rapid propagation.

1.1.9 Crack Arrest Fail Safe Structure - This is structure which is designed and fabricated such that unstable rapid propagation will be stopped within a continuous area of the structure prior to complete failure and the strength and safety of the remaining undamaged structure will not be degraded below a specified level for a specified period of unrepaired service usage.

1.1.10 Multiple Load Path-Fail Safe Structure - This is structure which is designed and fabricated in segments (with each segment consisting of one or more individual elements) such that failure of any single segment

(i.e. load path) will not degrade the strength and safety below a specified level for a specified period of unrepaired service usage.

1.1.10.1 Multiple Load Path-Dependent Structure - Mulitple load path structure is classified as dependent if, by design, a common source of cracking exists in adjacent load paths at one location due to the nature of the assembly or manufacturing procedures. An example of multiple load path-dependent structure is planked tension skin where individual members are spliced in the spanwise direction by common fasteners with common drilling and assembly operations.

1.1.10.2 Multiple Load Path-Independent Structure - Multiple load path structure is classified as independent if by design, it is unlikely that a common source of cracking exists in more than a single load path at one location due to the nature of assembly or manufacturing procedures.

1.2 General Requirements

1.2.1 Analysis Requirements - It shall be a requirement to classify all safety of flight structure with regard to type of damage tolerance approach and degree of inspectability and perform the required analytical work necessary to demonstrate compliance with specific requirements in this specification. The analysis shall assume the presence of crack-like defects, placed in the most unfavorable orientation with respect to the applied stress and the material properties, and shall predict the growth behavior in the chemical, thermal, and sustained and cyclic stress environment to which that portion of the component shall be subjected. In addition, the interaction effects of variable loading shall be considered. Regardless of

the damage tolerance concept single initial flaws of the specified size shall be assumed to exist in each separate element of the structure. For structural elements where it is likely due to the fabrication and assembly operations that the flaws in two or more elements exist at the same location in the structure this shall be assumed.

1.2.2 Residual Strength Requirements - The minimum required residual strength is the minimum load which must be sustained by the aircraft with damage present without endangering safety of flight or degrading the performance of the aircraft for the specified minimum period of unrepaired service usage. This includes loss of strength, loss of stiffness, excessive permanent deformation, loss of control, or by reduction of the flutter speed below V_L. The minimum residual strength requirements are specified in Sections 2.0 through 4.0 in terms of the minimum load P_{XX} that the structure must be able to sustain at any time during the specified minimum period of unrepaired service usage with the specified damage present. The magnitude of P_{XX} varies with the overall degree of inspectability of the structure (e.g. P_{FE} applies to flight evident, P_{SV} applies to special visual inspectable, etc). The P_{XX} load shall be determined from average load exceedance data and shall be that load that could occur once in 100 times the applicable inspection interval (e.g. P_{DM} is the load that could occur once in 100 depot or base level inspection intervals). For fail safe structure there is a requirement to sustain a minimum load, P_{YY}, at the instant

of load path failure (or crack arrest) in addition to being able to sustain the load, P_{XX}, subsequent to load path failure (or crack arrest) at any time during the specified interval. The single load path failure (or crack arrest) load, P_{YY}, shall include a dynamic factor (D.F.). In lieu of test or analytical data to the contrary a dynamic factor of 1.15 shall be used. The magnitude of P_{YY} shall depend upon the overall inspectability and the specific inspectability of the intact structure for subcritical damage (i.e. damage less than failed load paths or arrested cracks). P_{YY} shall be determined per the following table:

OVERALL DEGREE OF INSPECTABILITY	INSPECTABILITY FOR MIN. ASSUMED IN-SERVICE SUB-CRITICAL DAMAGE SIZES	P_{YY}
In-Flight Evident	Walkaround Visual	D.F. X P_{WV}
	Special Visual	D.F. X P_{SV}
	Depot or Base Level	D.F. X P_{DM}
	Non-Inspectable	D.F. X P_{LT}
Ground Evident	Walkaround Visual	D.F. X P_{WV}
	Special Visual	D.F. X P_{SV}
	Depot or Base Level	D.F. X P_{DM}
	Non-Inspectable	D.F. X P_{LT}
Walkaround Visual	Walkaround Visual	D.F. X P_{WV}
	Special Visual	D.F. X P_{SV}
	Depot or Base Level	D.F. X P_{DM}
	Non-Inspectable	D.F. X P_{LT}
Special Visual	Special Visual	D.F. X P_{SV}
	Depot or Base Level	D.F. X P_{DM}
	Non-Inspectable	D.F. X P_{LT}
Depot or Base Level	Depot or Base Level	D.F. X P_{DM}
	Non-Inspectable	D.F. X P_{LT}
Non-Inspectable	Non-Inspectable	D.F. X P_{LT}

1.2.3 Test Requirements

1.2.3.1 Specimen Testing - Valid data shall be determined in accordance with the procedures set forth in the 1970 ASTM Standards Test Method E3999-70T, or as described in AFFDL-TR-69-111 or by alternate methods approved by the procuring agency. The materials from which the structure identified in Paragraph 1.2.1 are to be fabricated shall be controlled by a system of procedures and/or specifications which are sufficient to preclude the utilization in fracture critical areas of materials possessing K_{IC} (or K_C) values inferior to those assumed in design. Tests will be conducted on all billets, forgings, extrusions, plates, or other forms (from which final parts are to be finished) to evaluate the fracture toughness. A slice will be cut from these items, or integral projections thereof, at receiving inspections, so that specimens from each slice may be tested. These specimens shall have been heat treated with the same material from which they were cut. When sufficient data are available, sampling procedures may be instituted on approval of the Air Force.

1.2.3.2 Component Testing - Fail safe tests will be conducted on that structure which is considered to be fail safe to verify that the failure of a load path or rapid propagation of a crack will not result in loss of the entire structure. Tests will be performed during the preproduction design verification component test program and the full scale qualification test program. These tests will be conducted by pre-cracking a particular member to the critical crack length and applying the load P_{YY}. Tests will be conducted on selected critical structure, particularly slow crack growth components, to verify the

analytical crack propagation rates. Initial flaws of the specified size will be initiated at the critical point(s) and propagation rates measured. These tests will be performed during the preproduction design verification test program and during the full scale qualification test program. Wherever possible, the structural components used for static test and fatigue test will be used to perform these tests. If in certain cases, this is not possible, then additional components will be fabricated for testing.

1.2.4 Fracture Control Plan. General guidelines for the fracture control plan are provided in 5.1.3 of MIL-STD-XXX.

2.0 Slow Crack Growth Structure

2.1 Walkaround Inspectable

2.1.1 The Frequency of Inspection and inspection interval shall be specified in the system RFP, Prime Item Development Specification or other contract documents as applicable.

2.1.2 The Minimum Period of Unrepaired Service Usage shall be five (5) times the inspection interval specified in 2.1.1.

2.1.3 The Minimum Required Residual Strength shall be P_{WV}.

2.1.4 Minimum Assumed Initial Damage. The damage assumed to exist in new structure as a result of fabrication operations shall be an .050" long through the thickness crack emanating from one side of a hole. At locations other than holes the assumed initial damage size shall be (a/Q) = .100 where a is measured in the principal direction of crack growth and Q is the flaw shape parameter. A smaller initial flaw size may be assumed subsequent to a demonstration that all flaws larger than this assumed size have at least a 90% probability of detection with a 95% confidence using the selected production inspection procedure, equipment and personnel. This demonstration shall be subject to USAF approval. A smaller initial size may be assumed if proof test inspection is used. In this case the minimum assumed initial size shall be the calculated critical size at the proof test stress levels and temperature using the upper bound of the material K_{IC} data.

2.1.5 Minimum Assumed in-Service Damage Size - The smallest damage which can be presumed to exist in fuel tank structure after completion of a walk-around inspection shall be an uncovered open 2" through the thickness crack. A smaller through the thickness crack may be assumed only after it is shown (analytically or experimentally) that fuel leakage will occur and can be detected during the inspection. Other slow crack growth structure shall be assumed to be walkaround uninspectable.

2.1.6 Damage Growth Limits.

2.1.6.1 Fabrication Damage - Initial damage as specified in Paragraph 2.1.4 shall not grow to critical size and cause failure of the structure due to the application of P_{DM} in the minimum period of unrepaired service usage specified in Paragraph 2.3.2.

2.1.6.2 In-Service Damage - In-service damage size specified in Paragraph 2.1.5 shall not grow to critical size and cause failure of the structure due to the application of P_{WV} in the minimum period of unrepaired service usage specified in Paragraph 2.1.2.

2.2 Special Visual Inspectable

2.2.1 The Frequency of Inspection and inspection intervals shall be specified in the system RFP, PIDS or other contract documents as applicable.

2.2.2 The Minimum Period of Unrepaired Service Usage shall be two (2) times the inspection interval specified in 2.2.1.

2.2.3 The Minimum Required Residual Strength shall be P_{SV}.

2.2.4 The Minimum Assumed Initial Damage shall be as specified in 2.1.4.

2.2.5 Minimum Assumed In-Service Damage Size - The smallest damage which can be presumed to exist in the structure after completion of special visual inspection shall be an uncovered open 2" through the thickness crack. A smaller through the thickness crack may be assumed only in those special cases where inspection statistics on similar structure or unique design features clearly indicate that smaller cracks can and will be found.

2.2.6 Damage Growth Limits.

2.2.6.1 Fabrication Damage - 2.1.6.1 applies.

2.2.6.2 In-Service Damage - In-service damage size specified in Paragraph 2.2.5 shall not grow to critical size and cause failure of the structure due to the application of P_{SV} in the minimum period of unrepaired service usage specified in Paragraph 2.2.2

2.3 Depot or Base Level Inspectable

2.3.1 The Frequency of Inspection and inspection intervals shall be specified in the system RFP, PIDS or other contract documents as applicable.

2.3.2 The Minimum Period of Unrepaired Service Usage shall be two (2) times the inspection interval specified in 2.3.1.

2.3.3 The Minimum Required Residual Strength shall be P_{DM}.

2.3.4 The Minimum Assumed Initial Damage shall be as specified in 2.1.4.

2.3.5 The Minimum Assumed In Service Damage Size - The smallest damage which can be presumed to exist in the structure after completion of a depot or base level inspection shall be as follows:

2.3.5.1 If The Component is to be removed from the aircraft and completely inspected with NDI procedures equivalent to those performed during fabrication, the minimum assumed damage size shall be that specified in 2.1.4.

2.3.5.2 Where NDI Techniques such as penetrant, magnetic particle or ultrasonics are applied to a component installed in the aircraft, the minimum assumed size shall be a through the thickness crack emanating from a fastener hole, having 0.250" of uncovered length. At other locations, the minimum assumed damage size shall be a/Q = 0.20".

2.3.5.3 Where Visual Inspection is used, a 2" uncovered open through the thickness crack shall be the minimum size.

2.3.5.4 Smaller Flaw Sizes may be assumed under Paragraphs 2.3.5.2 and 2.3.5.3 subsequent to a demonstration that all flaws larger than the selected size have at least a 90% probability of detection with a 95% confidence using the specified in-service inspection procedures and equipment. This demonstration shall be subject to USAF approval.

2.3.5.5 Smaller Flaw Sizes may be assumed under 2.3.5.2 and 2.3.5.3 if deopt or base level proof test inspection is used. In this case the minimum assumed sizes shall be calculated critical sizes at the proof test stress levels and temperatures using the upper bound of the material K_{IC} data.

2.3.6 Damage Growth Limits.

2.3.6.1 Fabrication Damage - 2.1.6.1 applies.

2.3.6.2 In-Service Damage - In-service damage size specified in Paragraph 2.3.5 shall not grow to critical size and cause failure of the structure due to the application of P_{DN} in the minimum period of unrepaired service usage specified in Paragraph 2.3.2.

2.4 Non-Inspectable

2.4.1 The Frequency of Inspection is not applicable

2.4.2 The Minimum Period of Unrepaired Service Usage shall be two (2) design service lifetimes.

2.4.3 The Minimum Required Residual Strength shall be P_{LT}.

2.4.4 The Minimum Assumed Initial Damage shall be as specified in 2.1.4.

2.4.5 The Minimum Assumed In-Service Damage Size is not applicable.

2.4.6 Damage Growth Limits - Initial damage as specified in Paragraph 2.1.4 shall not grow to critical size and cause failure of the structure due to the application of P_{LT} in the minimum period of unrepaired service usage as specified in Paragraph 2.4.2.

3.0 Fail Safe - Multiple Load Path (MLP) Structure

3.1 In-Flight Evident

3.1.1 Frequency of Inspection is not applicable

3.1.2 The Minimum Period of Unrepaired Service Usage shall be that period of time between that when the damage becomes evident and the completion of an immediate return to base.

3.1.3 The Minimum Required Residual Strength shall be P_{FE} subsequent to load path failure and P_{YY} at time of load path failure.

3.1.4 Minimum Assumed Initial Damage

3.1.4.1 Intact New Structure - The damage assumed to exist in each load path of new structure as a result of fabrication operations shall be an .020" long through the thickness crack emanating from one side of a hole. At locations other than holes the assumed initial damage sizes shall be $(a/Q) = .030''$ where a is measured in the principal direction of crack growth and Q is the flaw shape parameter. A smaller initial flaw size may be assumed subsequent to a demonstration that all flaws larger than this assumed size have at least a 90% probability of detection with a 50% confidence level using the selected production inspection procedure, equipment, and personnel. This demonstration shall be subject to USAF approval.

3.1.4.2 Remaining Structure at Time of And Subsequent to Load Path Failure - The damage assumed to exist adjacent to the primary failure in the remaining MLP dependent structure at time of and following the failure of a load path shall be euqal to an .020" long through the thickness crack emanating from one side of a hole or damage level equal to $(a/Q) = .030''$ at locations other than holes, plus the amount of growth Δ a which occurs

prior to load path failure. The damage assumed to exist adjacent to the primary failure in each load path of the remaining MLP independent structure at time of and following the failure of a load path shall be equal to an .010" radius semicircular corner crack emanating from one side of a hole or damage equal to (a/Q) = 0.010" at locations other than holes, plus the amount of growth Δa which occurs prior to load path failure.

3.1.5 The Minimum Assumed In-Service Damage Size shall be a failed load path.

3.1.6 Damage Growth Limits

3.1.6.1 Intact New Structure - Initial damage as specified in Paragraph 3.1.4.1 shall not grow to critical size and cause failure of a load path due to the application of P_{DM} in the minimum period of unrepaired service usage specified in Paragraph 3.5.2. If the structure is not inspectable for subcritical cracks, the initial damage specified in 3.1.4.1 shall not grow to critical size and cause failure of a load path due to the application of P_{LT} in one lifetime.

3.1.6.2 In Remaining Structure Subsequent to Load Path Failure - Damage as specified in Paragraph 3.1.4.2 shall not grow to critical size and cause failure of the remaining structure due to the application of P_{FE} in the minimum period of unrepaired service usage specified in Paragraph 3.1.2.

3.2 Ground Evident

3.2.1 Frequency of Inspection shall be once per flight.

3.2.2 The Minimum Period of Unrepaired Service Usage shall be one complete flight.

3.2.3 The Minimum Residual Strength shall be P_{GE} subsequent to load path failure and P_{YY} at time of load path failure.

3.2.4 Minimum Assumed Initial Damage

3.2.4.1 The Damage in Intact New Structure shall be as specified in 3.1.4.1.

3.2.4.2 The Damage in Remaining Structure at Time of And Subsequent to Load Path Failure - shall be as specified in 3.1.4.2.

3.2.5 The Minimum Assumed In-Service Damage Size shall be a failed load path.

3.2.6 Damage Growth Limits

3.2.6.1 Intact New Structure - 3.1.6.1 applies.

3.2.6.2 In Remaining Structure Subsequent to Load Path Failure - Damage as specified in Paragraph 3.1.4.2 shall not grow to critical size and cause failure of the remaining structure due to the application of P_{GE} in the minimum period of unrepaired service usage specified in Paragraph 3.2.2.

3.3 Walkaround Visual Inspectable

3.3.1 Frequency of Inspection and inspection interval shall be specified in the system RFP, PIDS or other contract document as applicable.

3.3.2 The Minimum Period of Unrepaired Service Usage shall be five (5) times the walkaround inspection interval specified in 3.3.1.

3.3.3 The Minimum Residual Strength shall be P_{WV} subsequent to in-service inspection, and P_{YY} at time of load path failure.

3.3.4 Minimum Assumed Initial Damage

3.3.4.1 The Damage in Intact New Structure shall be as specified in 3.1.4.1.

3.3.4.2 Remaining Structure Subsequent to Load Path Failure and Intact Structure Subsequent to In-Service Inspection - The damage assumed to exist adjacent to the primary failure in the remaining MLP dependent structure at time of and following the failure of a load path (or significant damage to the load path) shall be equal to an .020" long through the thickness crack emanating from one side of a hole or damage equal to a/Q = .030" at locations other than holes, plus the amount of growth Δ a which occurs prior to load path failure or prior to in-service inspection. The damage assumed to exist adjacent to the primary failure in each load path of the remaining MLP independent structure at time of and following failure of a load path (or significant damage to the load path) shall be euqal to an .010" radius semicircular corner crack emanating from one side of a hole or damage equal to a/Q = 0.010" at locations other than holes, plus the amount of growth Δ a which occurs prior to a load path failure or prior to in-service inspection.

3.3.5 The Minimum Assumed In-Service Damage shall be as specified in Paragraph 2.1.5 or a failed member, whichever is applicable.

3.3.6 Damage Growth Limits

3.3.6.1 Intact New Structure - 3.1.6.1 applies.

3.3.6.2 Intact Structure - Subsequent to In-Service Inspection - If the detectable damage is less than a failed load path then the minimum assumed damage in one load path shall be as specified in Paragraph 2.1.5. This damage plus the damage assumed to exist in the remaining structure at the time of inspection as defined in Paragraph 3.3.4.2, shall not grow

to critical size and cuase failure of the structure due to the application of P_{WV} in the minimum period of unrepaired service usage specified in Paragraph 3.3.2.

3.3.6.3 Remaining Structure - Subsequent to Load Path Failure - If the in-service detectable damage size is a failed load path then the damage in the remaining structure as defined in Paragraph 3.3.4.2 shall not grow to critical size and cause failure of the remaining structure due to the application of P_{WV} in the minimum period of unrepaired service usage specified in Paragraph 3.3.2.

3.4 Special Visual Inspectable

3.4.1 Frequency of Inspection and inspection intervals shall be specified in the systems RFP, PIDS or other contract document as applicable.

3.4.2 The Minimum Period of Unrepaired Service Usage shall be two (2) times the special visual inspection interval specified in 3.4.1.

3.4.3 The Minimum Required Residual Strength shall be P_{SV} subsequent to in-service inspection, and P_{YY} at time of load path failure.

3.4.4 The Minimum Assumed Initial Damage shall be as specified in Paragraph 3.3.4.

3.4.5 The Minimum Assumed In-Service Damage shall be as specified in 2.2.5 or a failed member, whichever is applicable.

3.4.6 Damage Growth Limits

3.4.6.1 Intact New Structure - 3.1.6.1 applies.

3.4.6.2 Intact Structure - Subsequent to In-Service Inspection - If the in-service detectable damage size is less than a failed load path then the

minimum assumed damage in one load path shall be as specified in Paragraph 2.2.4. This damage plus the damage assumed to exist in the remaining structure at the time of inspection as defined in Paragraph 3.3.4.2 shall not grow to critical size and cause failure of the structure due to the application of P_{SV} in the minimum period of unrepaired service usage as specified in Paragraph 3.4.2.

3.4.6.3 Remaining Structure - Subsequent to Load Path Failure - If the in-service detectable damage is a failed load path, then the damage in the remaining structure as defined in Paragraph 3.3.4.2 shall not grow to critical size and cause failure of the structure due to the application of P_{SV} in the minimum period of unrepaired service usage as specified in Paragraph 3.4.2.

3.5 Depot or Base Level Inspectable

3.5.1 The Frequency of Inspection and inspection interval shall be specified in the system RFP, PIDS, or other contract documents as applicable.

3.5.2 The Minimum Period of Unrepaired Service Usage shall be two (2) times the depot or base level inspection interval specified in 3.5.1.

3.5.3 The Minimum Residual Strength shall be P_{DM} subsequent to in-service inspection, and P_{YY} at time of load path failure.

3.5.4 Minimum Assumed Initial Damage shall be as specified in Paragraph 3.3.4.

3.5.5 The Minimum Assumed In-Service Damage shall be as specified in Paragraph 2.3.5 or a failed member whichever is applicable.

3.5.6 Damage Growth Limits

3.5.6.1 Intact New Structure - 3.1.6.1 applies.

3.5.6.2 Intact Structure - Subsequent to In-Service Inspection - If the in-service detectable damage is less than a failed load path, then the minimum assumed damage in one load path shall be as specified in 2.3.5. This damage plus the damage assumed to exist in the remaining structure at the time of inspection as defined in Paragraph 3.3.4.2 shall not grow to critical size and cause failure of the structure due to the application of P_{DM} in the minimum period of unrepaired service usage as specified in 3.5.2.

3.5.6.3 Remaining Structure - Subsequent to Load Path Failure - If the in-service detectable damage is a failed load path, then the damage in the remaining structure as defined in Paragraph 3.3.4.2 shall not grow to critical size and cause failure of the remaining structure due to the application of P_{DM} in the minimum period of unrepaired service usage specified in Paragraph 3.5.2.

3.6 In-Service Non-Inspectable

3.6.1 The Frequency of Inspection is not applicable.

3.6.2 The Minimum Period of Unrepaired Service Usage shall be one design service lifetime.

3.6.3 The Minimum Residual Strength shall be P_{LT} subsequent to load path failure, and P_{YY} at time of load path failure.

3.6.4 Minimum Assumed Initial Damage shall be as specified in Paragraph 3.3.4.

3.6.5 The Minimum Assumed In-Service Damage is not applicable.

3.6.6 Damage Growth Limits

3.6.6.1 Intact New Structure - Initial damage as specified in Paragraph 3.3.4.1 shall not grow to critical size and cause failure of a load path due to the application of P_{LT} in the minimum period of unrepaired service usage specified in Paragraph 3.6.2.

3.6.6.2 Remaining Structure - Subsequent to Load Path Failure - Subsequent to load path failure, initial damage in the remaining structure as specified in Paragraph 3.3.4.2 shall not grow to critical size and cause failure of the remaining structure due to the application of P_{LT} in the minimum period of unrepaired service usage specified in Paragraph 3.6.2.

4.0 Fail Safe - Crack Arrest Structure

4.1 In-Flight Evident

4.1.1 Frequency of Inspection is not applicable.

4.1.2 The Minimum Period of Unrepaired Service Usage shall be that period of time between that when the damage becomes evident and completion of an immediate return to base.

4.1.3 The Minimum Required Residual Strength shall be P_{YY} at time of crack arrest and P_{FE} subsequent to crack arrest.

4.1.4 Minimum Assumed Initial Damage.

4.1.4.1 The Damage in Intact New Structure shall be as specified in 3.1.4.1.

4.1.4.2 Remaining Structure At Time of and Subsequent to Crack Arrest. The damage assumed to exist in the remaining structure following arrest of a rapidly propagating crack shall depend upon the particular geometry. In conventional skin stringer (or frame) construction this shall be assumed as two panels (bays) of cracked skin plus the broken central stringer (or frame). Where tear straps are provided between stringers (or frames), this damage shall be assumed as cracked skin between tear straps plus the broken central stringer (or frame). Other configurations shall assume equivalent damage as mutually agreed upon by the contractor and the AF.

4.1.5 The Minimum Assumed In-Service Damage shall be as specified in Paragraph 4.1.4.2.

4.1.6 Damage Growth Limits

4.1.6.1 Intact New Structure - Initial damage as specified in Paragraph 3.1.4.1 shall not grow to the size which would cause an initial rapid propagation due to the application of P_{DM} in the minimum period of unrepaired service usage specified in Paragraph 4.5.2. If the structure is not inspectable for subcritical cracks, the initial damage specified in 3.1.4.1 shall not grow to the size which would cause an initial rapid crack propagation due to the application of P_{LT} in one lifetime.

4.1.6.2 Remaining Structure Subsequent to Crack Arrest - Damage as specified in Paragraph 4.1.4.2 shall not grow to the size required to cause complete structural failure due to the application of P_{FE} in the minimum period of unrepaired service usage specified in Paragraph 4.1.2.

4.2 Ground Evident

4.2.1 Frequency of Inspection shall be once per flight.

4.2.2 The Minimum Period of Unrepaired Service Usage shall be one complete flight.

4.2.3 The Minimum Required Residual Strength shall be P_{GE} subsequent to crack arrest and P_{YY} at time of crack arrest.

4.2.4 Minimum Assumed Initial Damage.

4.2.4.1 The Damage In Intact New Structure shall be as specified in 3.1.4.1.

4.2.4.2 The Damage In Remaining Structure at Time of And Subsequent to Crack Arrest shall be as specified in 4.1.4.2.

4.2.5 The Minimum Assumed In-Service Damage shall be as specified in Paragraph 4.1.4.2.

4.2.6 Damage Growth Limits.

4.2.6.1 Intact New Structure - 4.1.6.1 applies.

4.2.6.2 Remaining Structure Subsequent to Crack Arrest - Damage as specified in Paragraph 4.1.4.2 shall not grow to the size required to cause complete structural failure due to the application of P_{GE} in the minimum period of unrepaired service usage specified in Paragraph 4.2.2.

4.3 Walkaround Visual Inspectable

4.3.1 Frequency of Inspection shall be as specified in the system RFP, PIDS, or other contract documents as applicable.

4.3.2 The Minimum Period of Unrepaired Service Usage shall be five (5) times the walkaround inspection interval specified 4.3.1.

4.3.3 The Minimum Required Residual Strength shall be P_{WV} subsequent to in-service inspection, and P_{YY} at time of crack arrest.

4.3.4 Minimum Assumed Initial Damage

4.3.4.1 The Damage In Intact New Structure shall be as specified in 3.1.4.1.

4.3.4.2 The Damage In Remaining Structure at Time of And Subsequent to Crack Arrest shall be as specified in 4.1.4.2.

4.3.5 The Minimum Assumed In-Service Damage shall be as specified in Paragraph 2.1.5 (assumed to be located at an inaccessible, failed stringer or frame), or specified in Paragraph 4.1.4.2, whichever is applicable.

4.3.6 Damage Growth Limits

4.3.6.1 Intact New Structure - 4.1.6.2 applies.

4.3.6.2 Intact Structure - Subsequent to In-Service Inspection - If the in-service detectable damage is less than an arrested crack as deseribed in

Paragraph 4.1.4.2, then the minimum assumed damage as specified in Paragraph 4.3.5 shall not grow to the size required to cause complete structural failure due to the application of P_{WV} in the minimum period of unrepaired service usage specified in Paragraph 4.3.2.

4.3.6.3 Remaining Structure Subsequent to Crack Arrest - Damage as specified in Paragraph 4.3.5 shall not grow to the size required to cause complete structural failure due to the application of P_{WV} in the specified period of unrepaired usage specified in Paragraph 4.3.2.

4.4 Special Visual Inspectable

4.4.1 Frequency of Inspection shall be as specified in the system RFP, PIDS or other contract documents as applicable.

4.4.2 The Minimum Period of Unrepaired Service Usage shall be two (2) times the special visual inspection interval specified in 4.4.1.

4.4.3 The Minimum Required Residual Strength shall be P_{SV} subsequent to in-service inspection, and P_{YY} at time of crack arrest.

4.4.4 Minimum Assumed Initial Damage

4.4.4.1 The Damage In Intact New Structure shall be as specified in 3.1.4.1.

4.4.4.2 The Damage in Remaining Structure at Time of And Subsequent to Crack Arrest shall be as specified in 4.1.4.2.

4.4.5 The Minimum Assumed In-Service Damage shall be as specified in Paragraph 2.2.5 (assumed to be located at an inaccessible, failed stringer or frame), or as specified in Paragraph 4.1.4.2, whichever is applicable.

4.4.6 Damage Growth Limits

4.4.6.1 Intact New Structure - 4.1.6.2 applies.

4.4.6.2 Intact Structure - Subsequent to In-Service Inspection - If the in-service detectable damage is less than an arrested crack as described in Paragraph 4.1.4.2, then the minimum assumed damage as specified in Paragraph 4.4.5 shall not grow to the size required to cause complete structural failure due to the application of P_{SV} in the minimum period of unrepaired service usage specified in Paragraph 4.4.2.

4.4.6.3 Remaining Structure Subsequent to Crack Arrest - Damage as specified in Paragraph 4.1.4.2 shall not grow to the size required to cause complete structural failure due to the application of P_{SV} in the minimum period of unrepaired service usage specified in Paragraph 4.4.2.

4.5 Depot or Base Level Inspectable

4.5.1 Frequency of Inspection shall be specified in the system RFP, PIDS or other contract documents, as applicable.

4.5.2 The Minimum Period of Unrepaired Service Usage shall be two (2) times the depot or base level inspection interval specified in 4.5.1.

4.5.3 The Minimum Required Residual Strength shall be P_{DM} subsequent to in-service inspection and P_{YY} at time of crack arrest.

4.5.4 Minimum Assumed Initial Damage

4.5.4.1 The Damage In Intact New Structure shall be as specified in 3.1.4.1.

4.5.4.2 The Damage In Remaining Structure at Time of And Subsequent to Crack Arrest shall be as specified in 4.1.4.2.

4.5.5 The Minimum Assumed In-Service Damage shall be as specified in Paragraph 2.3.5 (assumed to be located at an inaccessible failed stringer or frame), or as specified in Paragraph 4.1.4.2, whichever is applicable.

4.5.6 Damage Growth Limits

4.5.6.1 Intact New Structure - 4.1.6.2 applies.

4.5.6.2 Intact Structure - Subsequent to In-Service Inspection - If the in-service detectable damage is less than an arrested crack as described in Paragraph 4.1.4.2, then the minimum assumed damage as specified in Paragraph 4.5.5 shall not grow to the size required to cause complete structural failure due to the application of P_{DM} in the minimum period of unrepaired service usage specified in Paragraph 4.5.2.

4.5.6.3 Remaining Structure Subsequent to Crack Arrest - Damage as specified in Paragraph 4.5.5 shall not grow to the size required to cause complete structural failure due to the application of P_{DM} in the minimum period of unrepaired service usage specified in Paragraph 4.5.2.

4.6 In-Service Non-Inspectable Crack Arrest Structure shall not be allowed.

SUPPLEMENT (F)

MIL-STD-1530

MIL-STD-1530 (USAF)
1 September 1972

MILITARY STANDARD

AIRCRAFT STRUCTURAL INTEGRITY PROGRAM, AIRPLANE REQUIREMENTS

FSC 1500

DEPARTMENT OF THE AIR FORCE

Aircraft Structural Integrity Program, Airplane Requirements

MIL-STD-1530 (USAF)

1. This Military Standard has been approved by the Department of the Air Force and is mandatory for use by activities under the cognizance of the Air Force effective as of date of issue.

2. Recommended corrections, additions, or deletions should be addressed to the Aeronautical Systems Division, 4950th Test Wing (TZSA), Wright-Patterson Air Force Base, Ohio 45433.

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TABLES

1. SCOPE

1.1 Purpose. The purpose of this standard is to define the requirements necessary to achieve structural integrity of USAF airplanes and to specify acceptance methods of contractor compliance. This standard shall be used by (a) contractors in conducting the development of an airframe for a particular weapon or support system; and (b) government personnel in managing the development, production, and operational support of a particular airplane system throughout its life cycle.

1.2 Applicability. This standard defines considerations required to ensure the structural integrity of airplane weapon and support systems. The degree of applicability of the various portions of this standard may vary between airplane systems as described in 1.3.

1.2.1 Applicability to type of aircraft system. This standard is directly applicable to power-driven aircraft having fixed or adjustable fixed wings and to those portions of manned helicopter and V/STOL aircraft which have similar structural characteristics. Helicopter-type power-transmission systems, including lifting and control rotors, and other dynamic machinery, and power generators, engines, and propulsion systems are not covered by this standard. For unmanned drones and remotely piloted vehicles, certain requirements of this standard may be waived or factors of safety reduced commensurate with sufficient structural safety and durability to meet the intended use of the airframe. Waivers and deviations shall be specified in the request for proposal or contract specifications or shall have specific Air Force approval prior to commitment in the design.

1.2.2 Applicability to type of program. This standard applies to (a) future airplane systems; (b) airplane systems procured by the Air Force but developed under the auspices of another regulatory activity (such as the FAA or USN); and (c) airplanes modified and directed to new missions.

1.3 Modifications. The Air Force will make the decision regarding application of this standard and may modify requirements of this standard to suit system needs. The description of the modifications shall be documented in accordance with 5.1.1 of this standard.

2. REFERENCED DOCUMENTS

2.1 The following documents, of the issue in effect on date of invitation for bids or request for proposal, form a part of this standard to the extent specified herein:

SPECIFICATIONS

MIL-I-6870	Inspection Requirements, Nondestructive, for Aircraft Materials and Parts
MIL-A-8860	Airplane Strength and Rigidity, General Specification for
MIL-A-8861	Airplane Strength and Rigidity, Flight Loads

MIL-A-8862	Airplane Strength and Rigidity, Landing and Ground Handling Loads
MIL-A-8865	Airplane Strength and Rigidity, Miscellaneous Loads
MIL-A-8866	Airplane Strength and Rigidity, Reliability Requirements, Repeated Loads, and Fatigue
MIL-A-8867	Airplane Strength and Rigidity, Ground Tests
MIL-A-8868	Airplane Strength and Rigidity, Data and Reports
MIL-A-8869	Airplane Strength and Rigidity, Nuclear Weapons Effects
MIL-A-8870	Airplane Strength and Rigidity, Flutter, Divergence and Other Aeroelastic Instabilities
MIL-A-8871	Airplane Strength and Rigidity, Flight and Ground Operations Tests
MIL-A-8892	Airplane Strength and Rigidity, Vibration
MIL-A-8893	Airplane Strength and Rigidity, Sonic Fatigue
MIL-R-83165	Recorder, Signal Data MXU-553/A
MIL-C-83166	Converter-Multiplexer, Signal Data, General Specification for

STANDARD

MIL-STD-882	System Safety Program for Systems and Associated Subsystems and Equipment, Requirements for

PUBLICATIONS

Military Handbooks

MIL-HDBK-5	Metallic Materials and Elements for Aerospace Vehicle Structures
MIL-HDBK-17	Plastics for Flight Vehicles
MIL-HDBK-23	Structural Sandwich Composites

Air Force Systems Command Design Handbooks

DH 1-0	General
DH 1-2	General Design Factors
DH 2-0	Aeronautical Systems
DH 2-7	System Survivability

Air Force Technical Order

00-25-4	Depot Level Maintenance, Aerospace Vehicle and Training Devices

(Copies of documents required by suppliers in connection with specific procurement functions should be obtained from the procuring activity or as directed by the contracting officer.)

3. DEFINITIONS. The definitions contained throughout this document or included in the documents listed in Section 2 shall apply.

4. GENERAL REQUIREMENTS

4.1 Discussion. The effectiveness of any military force depends in part on the operational readiness of weapon systems. One major item of an airplane system affecting its operational readiness is the condition of the airframe structure (including landing gear). To maintain operational readiness, the capabilities, condition, and operational limitations of the airframe of each airplane weapon and support system must be established. Any potential structural or material problems must be identified early in the life cycle to minimize their impact on the operational fleet, and a preventative maintenance program must be determined to provide for the orderly scheduling of inspections and replacement or repair of life-limited elements of the airframe.

4.1.1 The overall program to provide USAF airplanes with the required structural characteristics is referred to as the aircraft structural integrity program (ASIP) (reference table I). General requirements of the ASIP are to:

a. Establish, evaluate, and substantiate the structural integrity (airframe strength, rigidity, and service life) of airplane systems

b. Acquire, evaluate, and utilize operational usage data to provide a continual record of the in-service integrity of the aircraft

c. Provide a basis for determining logistics and force planning requirements (maintenance, inspections, supplies, rotation of aircraft, system phaseout, and future force structure)

d. Provide a basis to improve structural criteria and methods of design, evaluation, and substantiation for future airplane systems.

4.1.2 The majority of detail requirements are published in existing military specifications and will only be referenced in section 5. Those requirements which are not included elsewhere are contained herein. The Air Force shall

Table I. USAF Aircraft Structural Integrity Program Tasks

Task I	Task II	Task III	Task IV	Task V
Design Information	Design Analyses And Development Tests	Full Scale Testing	Fleet Management Data Package	Fleet Management
ASIP master plan	Material and joint allowables	Static test	Final analyses	Loads/environment spectra survey support
Structural design criteria	Loads analysis	Damage tolerance tests	Strength summary	Service monitoring program
Fracture and fatigue control plan	Temperature analysis	Fatigue tests	Parametric analysis	Service inspection maintenance and repair
Selection of materials, processes and joining methods	Stress analysis	Sonic fatigue tests	Instrumentation and data recording provisions	Structural performance records
Planned operational usage	Damage tolerance analysis	Flight and ground loads survey	Service inspection and maintenance control	
	Fatigue analysis	Flutter tests		
	Sonic fatigue analysis	Flight flutter tests		
	Vibration analysis	Loads/environment spectra survey		
	Flutter analysis			
	Nuclear weapons effects analysis			
	Nonnuclear weapons effects analysis			
	Design development and preproduction design verification tests			

resolve any differences in detail requirements that may exist between this standard and the referenced documents listed in section 2 herein during preparation of the request for proposal. Any differences discovered by the contractor shall be brought to the immediate attention of the Air Force. The applicable specifications, including the latest revisions thereto, for a particular airplane system shall be as stated in the request for proposal or the contract as appropriate. Air Force approval is required for all contractor actions involving system structural integrity.

4.2 Requirements. ASIP consists of the following five interrelated functional tasks (see 6.3):

a. Task I (design information): Development of those criteria which must be applied during design so that the specific requirements will be met.

b. Task II (design analysis and development tests): Determination of the environment in which the airframe must operate and survive based on the applied criteria, and the response of the airframe to the environment.

c. Task III (full-scale testing): Flight and laboratory tests of the airframe to determine the structural adequacy of the design.

d. Task IV (fleet management data package): Generation by the contractor of the data required to manage fleet operations in terms of inspections, modifications, and damage assessments.

e. Task V (fleet management): Those operations that must be conducted by the Air Force during fleet operations to ensure the safety and durability of the fleet throughout the useful life of the airplane.

5. DETAIL REQUIREMENTS

5.1 Design information (task I). The design information task encompasses those efforts required to apply the existing theoretical, experimental, applied research, and operational experience to specific criteria for materials selection and structural design for the airplane system. The objective is to ensure that the appropriate criteria and planned usage are applied to an airplane system so that the specific operational requirements will be met. This task begins as early as possible in the conceptual phase and is finalized in subsequent phases of the system life cycle.

5.1.1 ASIP master plan. Detail requirements for an ASIP master plan shall be specified in the request for proposal. The master plan submitted by the contractor shall be approved by the Air Force. The master plan shall be prepared to develop a specific approach for accomplishment of the various ASIP tasks throughout the life cycle of the aircraft. This plan shall be provided as a part of the response to the request for proposal for each weapon system. Included in the master plan shall be the aircraft service life requirements as well as a detailed ASIP data flow diagram which assigns specific responsibilities. The plan shall depict the integration of the required ASIP elements into a logical sequence including time phased scheduling of all tasks for design development and qualification of the airplane structure. The required elements of the ASIP are defined in section 5. Discussion shall be provided showing how the plan was developed, its unique features, exceptions to the requirements of this standard, and any problems anticipated in executing the plan. The discussion of exceptions shall include complete justification and impact. A discussion of the development of the schedule shall also be provided, especially on interfaces, impact of schedule delays (e.g., from test failure), mechanisms for recovery programming, and other problem areas. The schedule shall be updated by the contractor on a continuing basis.

5.1.2 Structural design criteria. Detail structural design criteria for the specific airplane system shall be established by the contractor in accordance with the requirements of the specifications listed in 5.1.2.2. In addition, a brief description of the service-life criteria is contained in 5.1.2.1 since new requirements for damage tolerant design of USAF systems are being levied with the publication of this standard. The detail damage tolerant design requirements are specified in MIL-A-8866.

5.1.2.1 Service-life design criteria. The aircraft structure shall incorporate materials, stress levels, and structural configurations which (a) allow routine in-service inspection; (b) minimize crack initiation; and (c) minimize the probability of loss of the aircraft due to propagation of undetected fatigue cracks, flaws, or other damage. Durable structural designs which are resistant to crack initiation shall be a primary requirement to achieve Air Force weapon and support systems with low structural maintenance needs. Design life requirements shall be as specified in table II unless otherwise specified in the request for proposal or the contract specification. In addition, damage tolerant design shall be required for primary structure to ensure structural safety since undetected flaws or damage can and sometimes do exist in critical structural components despite the design, fabrication, and inspection efforts expended to eliminate their occurrence.

Table II. Life Requirements for Aircraft Structures (1)

	Years of Service	Flight Hours	Number of Flights	Landings (2)	Fuselage Pressurizations
Fighter					
Air superiority					
Long range	15	8,000	6,000	8,000	8,000
Short range	15	6,000	8,000	10,000	8,000
Ground attack	15	8,000	8,000	10,000	8,000
Bomber	25	15,000	3,000	5,000	5,000
Tanker	25	20,000	5,000	7,500	7,500
Cargo (3)					
Medium & heavy	25	50,000	12,500	25,000	15,000
Assault	25	15,000	12,500	20,000	15,000
Utility	25	25,000	15,000	20,000	20,000
AEW&C (4)	20	40,000	4,000	8,000	6,000
Trainer					
Primary	25	15,000	15,000	40,000	15,000
Navigational	25	25,000	6,000	10,000	7,500

NOTES:
(1) This table constitutes minimum structural design criteria and should not be used to interpret operational use (such as hours per flight)
(2) Full stop landings are assumed equivalent to the number of flights. Remainder are touch and goes
(3) Includes STOL & VTOL
(4) Includes command post systems

5.1.2.1.1 Durability. Structural durability requires that the areas of the structure that could be susceptible to fatigue, corrosion, or other crack initiation mechanisms be identified by analyses and tests (including material, component, development, and full-scale tests). This shall be accomplished in accordance with this standard, MIL-A-8866, and MIL-A-8867. Substantiated modifications and preventative measures as may be required to achieve low-maintenance service life shall be incorporated by the contractor prior to airplane delivery or as retrofit installation in fleet airplanes.

5.1.2.1.2 Safety. Damage tolerant design concepts can be categorized into the following two general categories: (a) those where unstable crack propagation is locally contained through the use of multiple load paths and/or tear stoppers (i.e., these concepts are referred to as fail-safe), and (b) those where flaws or defects are not allowed to attain the critical size required for unstable rapid propagation (i.e., these are referred to as slow-crack-growth concepts). Both design approaches shall assume the presence of initial damage (i.e., undetected flaws and defects) and shall have a specified minimum residual strength level both during and at the end of a specified period of unrepaired service usage. The initial damage size assumptions, damage growth limits, residual strength requirements and the minimum periods of uprepaired service usage depend on the type of structure and the damage detectability. Compliance with the specific damage tolerance requirements specified in MIL-A-8866 shall be required.

5.1.2.2 Structural design criteria requirements. Using the requirements specified in the systems specifications and the referenced Military specifications and standards, the contractor shall prepare the detailed structural design criteria for the particular airplane system. These criteria and all elements thereof shall be subject to Air Force approval. Detail structural design criteria are contained in AFSC DH 1-0 and DH 2-0 and in MIL-A-8860, MIL-A-8861, MIL-A-8862, MIL-A-8865, MIL-A-8866, MIL-A-8869, MIL-A-8870, MIL-A-8892, and MIL-A-8893. In addition, specific battle-damage criteria shall be provided by the Air Force in the request for proposal or contract, as appropriate, for each new system that is designed to operate in a hostile environment. These criteria shall include such items as threat, flight conditions, load-carrying capability, and duration after damage is imposed. The structure shall be designed to these criteria along with other specified requirements of AFSC DH 2-7.

5.1.3 Fracture and fatigue control plan. The contractor shall prepare a fracture and fatigue control plan and conduct the resulting program in accordance with this standard, MIL-A-8866, and MIL-A-8867. The overall plan plus the specified individual elements thereof shall be approved by the Air Force. The purpose of this plan is to identify and specifically define all of those tasks necessary to ensure compliance with the design service-life criteria specified

in 5.1.2.1 and the detail damage tolerance and fatigue requirements of MIL-A-8866. The objectives of the fracture and fatigue control program are to minimize service maintenance problems due to fatigue and other crack initiation mechanisms and to prevent the failure of safety-of-flight structure. The contractor shall prepare such a plan, obtain Air Force approval of the plan, and conduct the program in accordance with the plan. While many of the tasks in a comprehensive fracture and fatigue control program have been normal to past aircraft development efforts, the new and importantly different service-life criteria imposes the need for new tasks as well as tighter controls and more interdisciplinary involvement in the conventional tasks. The disciplines of fracture mechanics, fatigue, materials and processes, structural analysis, loads analysis, design, manufacturing, quality control, and nondestructive inspection are all intimately involved in fracture and fatigue control. Detail requirements for the fatigue control program are included in MIL-A-8866. The more important new requirements being levied as a result of the new service-life criteria are as follows:

a. Damage tolerance design concept/material/weight/cost trade studies shall be performed during the early design phases to obtain low-weight, cost-effective designs which comply with the requirements of MIL-A-8866.

b. Basic fracture data (i.e., K_{IC}, K_C, K_{ISCC}, da/dN, et cetera) shall be obtained from existing sources or developed as part of the contract to support the initial trade studies and the final design and analysis.

c. A criteria for identifying all fracture-critical parts shall be established by the contractor and approved by the Air Force.

d. A fracture and fatigue critical parts list shall be established and shall be kept current as the design of the airframe progresses.

e. Design drawings for the fracture and fatigue critical parts shall be zoned to identify critical locations within the part or assembly; define the acceptance limits or defect size, location, and orientation; and reference inspection procedures to be used.

f. Complete nondestructive inspection requirements, process-control requirements, corrosion-control requirements, quality control requirements, and fatigue-control requirements for all fracture and fatigue critical parts shall be established by the contractor and approved by the Air Force. Nondestructive inspections shall comply with MIL-I-6870. This effort shall include the proposed plan for certifying and monitoring subcontractor, vendor, and supplier inspection and quality control.

g. Where the designs are based on initial flaw-size assumptions less than those specified in MIL-A-8866 or the contract specifications, a nondestructive inspection demonstration program shall be performed by the contractor and approved by the Air Force to verify that all flaws equal to or greater than the design flaw sizes will be detected to the reliability and confidence levels specified in MIL-A-8866. Specifications of these inspection techniques shall become the manufacturing inspections requirements and may not be significantly changed without a requalifying demonstration program.

h. Material procurement and manufacturing process specifications shall be updated as necessary to ensure that basic materials and the resulting fracture and fatigue critical parts do not have fracture-toughness properties in the important loading directions which are less than those used in design.

i. Materials used in fracture and fatigue critical parts shall have traceability.

j. Damage-tolerance and fatigue analyses, development testing, and proof-of-compliance testing shall be performed in accordance with this standard, MIL-A-8866, and MIL-A-8867.

k. Detailed inspection, maintenance, and test procedures shall be developed for use on all fracture and fatigue critical parts during the scheduled depot-level or special field-service inspections. The request for proposal and the contract specifications shall specify the inspection intervals to be used in the design of the aircraft.

5.1.4 Selection of materials, processes, and joining methods. The objective in selecting materials, processes, and joining methods for primary airframe structure shall be to select those which will result in an efficient structure (i.e., a light-weight, cost-effective structure) that meets the strength, corrosion, and service-life requirements of this standard and supporting specifications. The selection of materials and processes, including joining for tension-loaded primary structure, shall be dominantly controlled by weight, corrosion factor, cost, and service life considerations. A primary factor in the final selection shall be the results of the design concept/material/weight/cost trade studies performed as an initial part of the fracture and fatigue control program. For structure which is primarily designed for compression strength, stiffness, or other considerations, material properties other than fatigue and fracture will normally govern the final selection. However, even in these cases fatigue can be an important consideration particularly at joints.

5.1.4.1 Structural materials, processes, and joining methods selection requirements. In response to the request for proposal, prospective contractors shall identify the proposed materials, processes, and joining methods to be used in each of the primary structural components and the rationale for the individual selections. After contract award and during the design phase, the contractor shall document the complete rationale used in the final selection for each

primary structural component. This rationale shall include all pertinent data upon which the selections were based including the data base, previous experience, and trade study results. All requirements of AFSC DH 1-2, sections 7A, "Materials," and 7B, "Processes," shall be met as applicable. All materials, processes, and joining method selections for fracture and fatigue critical parts shall be subject to Air Force approval.

5.1.5 Planned operational usage. Development of planned operational usage starts with the concept of the airplane system. The objective is to obtain information for use with statistical data previously obtained from flight maneuvers, turbulence, and ground-loading conditions encountered in operational use of similar aircraft to provide a definitive basis for deriving technically sound structural design requirements and fatigue loading/environment spectra. During the conceptual phase of the airplane system, maximum effort will be exerted by the Air Force to identify possible usages as envisioned by advanced planning activities (Hq USAF, Hq AFSC, and using commands). This is required to give early consideration to the loads and conditions of use resulting from flight during selected missions, landing impact, and ground operations. Table II provides the minimum service life requirements for various types of airplanes. Unless otherwise specified in the request for proposal, the requirements of table II shall apply.

5.1.5.1 Planned operational usage requirements. The detail requirements for deriving planned operational usage are contained in MIL-A-8866. The Air Force shall provide the planned usage and mission profiles (based on inputs from the using commands) as part of the request for proposal. These data shall be used as a basis for the contractor's proposal responses and shall be used in the initial design and analysis of the airframe. All revisions in these data subsequent to contract negotiations shall be at the discretion of the Air Force and, as deemed necessary, will be imposed on the contractor as a contract change.

5.2 Design analyses and development tests (task II). The objective of the design analyses and development test task is to determine the environment in which the structure will operate and survive (load, temperature, moisture, corrosion, abrasive environment, acoustic excitation), the response of the structure to these environments, and the service life based upon the planned operational usage and associated environment. Data developed in this task constitutes part of a structural data package which is the property of the Air Force and will be used by the Air Force to supplement the existing technology base (see 5.1).

5.2.1 Material and joint allowables. An integral part of the static strength, damage tolerance, and fatigue analyses are the material and joint allowables used in the analyses. The contractor shall utilize, as appropriate, the materials and joint allowables data in MIL-HDBK-5, MIL-HDBK-17, MIL-HDBK-23, and the damage tolerance section of AFSC DH 2-0. Other data sources may also be used subject to the approval of the Air Force. For those cases where there is insufficient data available, the contractor shall formulate and perform experimental programs to obtain the data. These programs shall be subject to Air Force approval.

5.2.1.1 Material fracture data. The primary reference for the material fracture data (i.e. K_{IC}, K_C, K_{ISCC}, da/dN, etc) to be used in damage tolerance analyses shall be the damage tolerance section of AFSC DH 2-0. It is unlikely that this reference will contain all of the needed data and as a result, an experimental and data collection program will be required to fill the gaps. The contractor shall formulate the needed program to obtain the data per the handbook guidelines.

5.2.1.2 Fatigue data. Fatigue data derived from constant-load amplitude-cyclic tests are often used for material and joint comparisons for selection of materials and processes. Ample evidence now exists that these comparisons may be suspect for use in the real-life variable load amplitude environment. Fundamental reasons for discrepancies are well understood qualitatively if not quantitatively. Therefore, applicable constant load amplitude fatigue data in MIL-HDBK-5 and MIL-HDBK-17 may be used for material and process selections and design of those critical parts exposed to a substantially constant-cyclic-loading environment throughout their lifetime. Where specifically applicable fatigue data is not available for the material condition and processes for joints, programs shall be defined to generate the required data. While constant-load amplitude data may be useful for initial screening and preliminary rough sizing, final material and process selection and final design of fatigue-critical parts and joints which are subject to a spectrum of variable load amplitudes in service shall utilize specific data from realistically representative spectrum fatigue tests. Since these data, in general, do not exist in handbooks or available literature, special test programs shall be defined to generate spectrum-type fatigue allowables for the specific materials and joining methods of fatigue critical primary structure.

5.2.2 Loads analysis. The contractor shall perform a loads analysis in accordance with the detail requirements specified in the request for proposal. The analysis shall be subject to Air Force approval. The loads analysis shall consist of determining the magnitude and distribution of all significant static and dynamic loads which the airplane is likely to encounter in performing its mission. This analysis consists of determination of inertia loads, aerodynamic loads, ground loads, powerplant loads, nuclear-weapons effects, sonic-fatigue loads, and repeated load spectra. Where applicable, this analysis shall include the effects of temperature, aeroelasticity, and dynamic response of the aircraft. The loads analysis for the various environments shall be revised as appropriate to incorporate the results of ground and flight tests.

5.2.3 Temperature analysis. The contractor shall perform a temperature analysis in accordance with the detail requirements specified in the request for proposal. This analysis shall be subject to Air Force approval. The analysis shall consist of determining the steady state and transient temperature distributions which the airplane is likely to encounter in performing its mission. Results of this analysis shall be combined with the loads and stress analyses to produce critical airplane temperature and heating rates (including thermal cycling) for use in design and testing of the airplane.

5.2.4 Stress analyses. The contractor shall perform a stress analysis in accordance with the detail requirements specified in the request for proposal. This analysis shall be subject to Air Force approval. The stress analysis shall consist of the analytical determination of the stresses, deformations, and margins of safety resulting from the external loads and temperature imposed on the airframe. The ability of the airplane structure to support the critical loads and to meet the specified strength requirements shall be established. In addition to verification of strength, the stress analysis shall be used as a basis for fatigue and fracture analyses, selection of critical structural components for preproduction tests, and selection of loading conditions to be used in the structural testing. The stress analysis shall also be used as a basis to determine the adequacy of structural changes throughout the life of the airplane and to determine the adequacy of the structure for new loading conditions that result from increased performance or new mission requirement. The stress distribution of the major components as determined by analysis shall be corrected as appropriate based on data obtained from structural tests. The analysis shall also be revised to reflect any major changes to the structure or to the loading conditions applied to the structure.

5.2.5 Damage tolerance analysis. The contractor shall perform a damage tolerance analysis in accordance with the detail requirements of MIL-A-8866 and as specified in the request for proposal. This analysis shall be approved by the Air Force. The analysis shall consist of determining the damage tolerance characteristics of the airframe. The objective is to substantiate the ability of the primary structural components to meet the specified residual strength, rigidity, and life requirements in the presence of specified initial flaws, battle damage, fatigue cracks, et cetera.

5.2.5.1 Analysis procedures. A flight-by-flight real time design load/environment spectrum shall be used in the crack-growth analysis. The initial calculations of critical flaw sizes, residual strengths, safe crack-growth lives, and inspection intervals shall use existing fracture test data and basic fracture-allowable data generated as part of the development program. The analysis shall be updated based on the results of design development tests, preproduction design verification tests, and any changes in the projected usage of the airplane. The effect of variability in fracture properties on the analytical results shall be accounted for in the damage tolerant design.

5.2.6 Fatigue analysis. The contractor shall perform a fatigue analysis in accordance with the detail requirements of MIL-A-8866 and the request for proposal. This analysis shall be subject to Air Force approval. The analysis shall consist of determining the resistance of the structure to fatigue cracking due to repeated application of load/environment. The objective of the fatigue

analysis is to determine the ability of the structure to provide the design service life (including scatter factor) as required in the request for proposal or procurement specifications. The fatigue analysis shall be performed for each new airplane design and for each subsequent series where there is a significant change in the structural configuration or loads. The analysis shall also be revised when there is additional information available from subsequent ground or flight tests or when a significant change occurs in the planned usage for a particular system. For proof of compliance, the contractor shall demonstrate resistance to fatigue cracking during the required design service life times a factor of 4 as specified in 5.3.3. For the purpose of fatigue analysis, variable scatter factors larger than 4 may be required for specific types of materials and structures. The contractor shall review applicable service experience, research results, and engineering test data to determine the specific factors required to meet the design objectives with a high probability of success. In addition to achieving fatigue-resistant design, the fatigue analysis shall provide a basis for development of test load spectra to be used in the design development, preproduction design verification, and full-scale fatigue tests.

5.2.7 Sonic fatigue analysis. The contractor shall perform a sonic fatigue analysis in accordance with the detail requirements of MIL-A-8893 and the request for proposal. This analysis shall be subject to Air Force approval. The objective of the sonic fatigue analysis is to ensure that the structure is resistant to sonic fatigue cracking for the design service life of the structure. The analysis shall define the intensity of the acoustic environment from all potentially critical sources and shall determine the dynamic stress response, including any significant thermal effects, and verify that the design service life requirements of MIL-A-8893 are met. Potentially critical sources include but are not limited to power plant noise, aerodynamic noise in regions of the turbulent and separated flow, exposed cavity resonance, and localized vibratory forces. The analysis shall be updated for any significant results obtained from the required laboratory and flight tests.

5.2.8 Vibration analysis. The contractor shall perform a vibration analysis in accordance with the detail requirements of MIL-A-8892 and the request for proposal. This analysis shall be subject to Air Force approval. The contractor's design shall control the structural vibration environment and predict the resultant environment in terms of vibration levels in various areas of the airplane such as the crew compartment, cargo areas, equipment bays, et cetera. The structure in each of these areas shall be resistant to fatigue cracking due to vibratory loads for the design service life. In addition, the airframe design shall control the vibration levels to that necessary for the reliable performance of personnel and equipment throughout the life of the airplane.

5.2.9 Flutter analysis. The contractor shall perform a flutter and divergence analysis in accordance with the detail requirements of MIL-A-8870 and the request for proposal. This analysis shall be subject to Air Force approval. The analysis shall consist of determination of the airplane flutter and divergence characteristics resulting from the interaction of the aerodynamic, inertia, and elastic characteristics of the components involved. The objective of the analysis is to substantiate the ability of both the damaged and undamaged airplane structure to meet the specified flutter and divergence margins. If significant differences in the aerodynamic, inertia, or elastic characteristics occur during design, design development testing, or are discovered during testing of the airplane or its components, the flutter and divergence analysis shall be revised accordingly.

5.2.10 Nuclear weapons effects analyses. The contractor shall perform a nuclear weapons effects analyses in accordance with the detail requirements of AFSC DH 2-7, MIL-A-8869, and the request for proposal. These analyses shall be subject to Air Force approval. Aircraft required to operate in nuclear environments can be exposed to transient radiation, thermal pulses, and pressure impulse loads from the explosion fields. The objectives of the nuclear weapons effects analyses are to: (a) verify that the design of the airframe structure will successfully resist the specified environmental conditions with no more than the specified residual damage; and (b) determine the structural capability envelope and crew radiation protection envelope for other degrees of survivability (damage) as may be required. The contractor shall prepare detail design criteria and shall conduct the nuclear weapons effects analyses for transient thermal, overpressure, and gust loads and provide the substantiation of allowable structural limits on those structures critical for these conditions. He shall also prepare and report the nuclear weapons effects capability envelope, including crew radiation protection, for a specified range of variations of weapon delivery trajectories, weapon size, aircraft escape maneuvers, and the resulting damage limits.

5.2.11 Nonnuclear weapons effects analysis. The contractor shall perform a nonnuclear weapon effects analysis in accordance with the criteria and requirements of AFSC DH 2-7 and the request for proposal. This analysis shall be subject to Air Force approval. Aircraft required to operate in hostile environments can be exposed to battle damage from a spectrum of threat capabilities. Damage-tolerance design capability has increased the potential for efficiently designing specific damage containment features into the structure to reduce the hazards of loss of aircraft due to small rocket and gunfire. The objective of the nonnuclear weapons effects analysis is to verify that the structural damage containment capability required to resist the specified threats does, in fact, exist.

5.2.12 Design-development and preproduction design-verification tests. These tests shall be conducted on structural elements and components during the design analysis task and are necessary to develop structural design concepts which meet the strength, stability, fatigue, damage tolerance, and inspectability requirements. The objective is to provide a realistic basis for the design analysis and full-scale structural tests.

5.2.12.1 Design-development tests. The design development tests shall be conducted early to establish the adequacy of basic design concepts, material selection, and configuration such as panel sizes, splices, and fittings. These tests will provide early evaluation of the design and analysis methods and structural properties of the proposed design and shall include structural configuration development tests for strength, fatigue, and damage-tolerance evaluation; tests of splices and joints including installation tolerances and hole limits; tests of panels, both plain and with cutouts; and tests of fittings and assemblies.

5.2.12.2 Preproduction design-verification tests. Preproduction design-verification tests shall be conducted to provide necessary design information to achieve a high degree of confidence in the strength, fatigue, and damage-tolerance properties of the design. Tests shall be conducted on assemblies and components selected from the critical areas using the earliest available production-type parts including forgings. However, prudent use of substitute parts for forgings may be necessary to ensure early test completion. These development tests are separate from the major structural tests which are conducted during the full-scale testing phase. The test spectra for the repeated load tests of major assemblies shall be based on the design spectrum. The appropriate test methods as discussed in 5.3.2 shall be used. These tests shall be conducted before heavy commitments are made to substantial quantities of production hardware and prior to the full-scale structural tests. The scheduling of these tests shall be such that they will not delay the full-scale structural tests. The preproduction design-verification tests shall include static, fatigue, and damage-tolerance tests of full-scale components and portions of major assemblies such as wing carry-through, horizontal tail support, wing pivots, crew compartment, landing gear, and support structure. These tests shall be conducted to verify the damage-tolerance characteristics of the critical primary components after the normal strength and fatigue requirements have been successfully demonstrated. Determination of the damage-tolerance characteristics of the critical components during these early tests shall provide information for input to the full-scale tests of the production airframe. Specific values of load level and damage size to be used during residual-strength and crack-growth tests shall be as specified or as determined by the contractor and approved by the Air Force.

5.2.12.3 Design-development and preproduction design-verification test requirements. Detail requirements for the design-development and preproduction design-verification tests are contained in MIL-A-8867. Prospective contractors shall estimate the scope of their proposed test plans in their response to the request for proposal. After contract award and during the initial design and analysis task, the contractor(s) shall finalize the plan and submit it to the Air Force for approval. The contractor shall revise and maintain approved updated versions of the test plan as the design develops. This plan shall provide the Air Force with the information needed to evaluate program adequacy. The plan shall consist of such information as rationale for selection and scope of tests, description of procedures, test loads and test factors, and analyses directed at establishing cost and schedule trade-offs used to develop the program.

5.3 Full-scale testing (task III). The objective of this task is to verify the structural integrity of the basic design and any necessary modifications through a series of ground and flight tests. Data developed in this task constitutes part of a structural data package which is the property of the Air Force and will be used by the Air Force to supplement the existing technology base (see 5.1).

5.3.1 Static tests. The contractor shall comply with the detail requirements for static tests specified in MIL-A-8867. Prior to initiation of testing, the test plans, procedures, and schedules shall be approved by the Air Force. The static test program shall consist of a series of laboratory tests, each conducted to at least 100 percent of design ultimate load on an instrumented airframe. These tests shall simulate the loads resulting from critical flight and ground handling conditions. Thermal environment effects shall be simulated along with load application on airframes where operational environments impose significant thermal effects. The primary purpose of the static test program is to verify the design ultimate static strength capabilities of the airframe. The results of failure load static tests shall also provide an indication of growth potential of the airframe and areas of excessive stress and/or deformations which could lead to in-service problems if not corrected.

5.3.1.1 Test article and test conditions. The static test airframe shall be a complete airframe assembly (including landing gear components) and shall be the first airframe constructed unless otherwise agreed to by the Air Force. Upon agreement by the Air Force, individual components (such as wing, empennage, fuselage, et cetera) may be tested separately if sufficient overlap of attaching structure is used to ensure proper load interactions at the structural interface. The static test airframe shall be tested to the design ultimate loads for critical conditions in accordance with MIL-A-8867. These critical conditions shall be defined by the contractor and approved by the Air Force prior to starting the test program. Intentional failing load tests, conducted at the completion of the design ultimate load tests, shall normally consist of one test for each major component (wing, fuselage, tail surfaces) and shall be negotiated in detail with the Air Force.

5.3.1.2 Test program scheduling. The static test, together with the combined flight load survey and structural integrity flight demonstration, shall be used to verify the static structural integrity of the airplane for the design limit flight envelope. As such, the static test program shall be so scheduled that no delays will be incurred in obtaining release for the structural flight test to 100 percent limit-load flight conditions.

5.3.2 Damage tolerance tests. The contractor shall comply with the detail requirements for damage tolerance tests as specified in MIL-A-8867. Prior to initiation of testing, the test plans, procedures, and schedules shall be approved by the Air Force. Damage tolerance tests (crack growth, residual strength) shall be performed on the full-scale static and fatigue test airframes at the conclusion of strength and failure test evaluations described in 5.3.1 and 5.3.3. These damage tolerance tests shall augment the tests conducted during the design-development and preproduction design-verification test programs and shall be structured to provide assurance of the safety of the airframe when exposed to initial flaws, fatigue cracks, battle damage, et cetera.

5.3.3 Fatigue tests. The contractor shall comply with the detail requirements for fatigue tests specified in MIL-A-8867. Prior to initiation of testing, the test plans, procedures, and schedules shall be approved by the Air Force. Fatigue tests of the airframe shall consist of repeated application of the spectrum of cyclic load/thermal environment simulating actual flight vehicle usage in realistic flight-by-flight sequence of loading (and heating/cooling as appropriate). These tests shall be based on the loading spectra and service-life requirements in the procurement specification. The tests shall be conducted to determine that the structural design has the capability to withstand the design service life times a factor of 4 without fatigue cracking. Fatigue cracks which initiate from fundamental design deficiencies prior to the specified test time shall require modifications. As in the case of initial structure, these modifications (and replacement parts for other than design deficiencies) shall also be tested to demonstrate the capability to sustain the full required factored lifetimes without fatigue cracking. Failure of a part as a result of causes other than fundamental design deficiences shall be exempt from design modifications only upon demonstration that adequate steps have been taken to preclude reoccurrence of the cause in delivered aircraft. The results of these tests, in association with other pertinent data, shall be the basis for determining fleet service utilization, inspection, and maintenance (see 5.4).

5.3.3.1 Test article. The test article shall be a complete basic airframe with no previous flight or test history and shall include all necessary landing gear components. Upon agreement by the Air Force, major components (such as wing, fuselage, empennage, et cetera) may be tested separately if sufficient overlap of attaching structure is used to ensure proper load interactions at the structural interface. The fatigue test article shall be representative of the

operational configuration. Prior to starting the fatigue test, all structural modifications required as a result of failures of preproduction design-verification component tests and the full-scale static test shall be incorporated in the test article unless otherwise agreed to by the Air Force.

5.3.3.2 Test program scheduling. An important requirement in the fatigue test program shall be that it be completed at the earliest possible time consistent with the desire to have an operational configuration. This is required to minimize fleet modification due to design deficiencies found during testing. To this end, the following shall be accomplished; (a) timely formulation of the test load spectrum; (b) early delivery of the test article; and (c) early establishment of management and contractual procedures for minimizing downtime in the event of test failure. Truncation of the design flight-by-flight spectrum for the purpose of shortening the test time shall be allowed only if it is demonstrated by separate comparative tests that the fatigue results of the foreshortened test spectrum are equivalent to the full flight by flight-test spectrum. The test equivalent damage shall precede the fleet by at least a factor of 4 based on average usage. It is also desirable that the test remain ahead of the highest usage airplane by a factor of 4.

5.3.3.3 Test program duration. For the purpose of contractual compliance, the fatigue test shall be considered completed at the end of four times the design lifetime. However, it may be advantageous to the Air Force to continue testing beyond the contractual requirement. This decision shall be made based upon the results of a joint review by the appropriate Air Force activities sufficiently prior to the scheduled test completion to allow continued testing should this be the decision. For the purpose of Air Force budgetary planning, the contractor, in his response to the request for proposal, shall provide an estimate of the additional incremental costs associated with each additional lifetime of testing.

5.3.3.4 Test program inspections. The fatigue test program shall contain the initial inspection, all repetitive inspection intervals, inspection methods, instrumentation, and techniques (specifications) devised from the design damage tolerance analysis (see 5.2.5) and as written into the inspection and maintenance technical orders (see 5.4.5). The objectives shall be to serve the dual purpose of closely monitoring the fatigue test structure for early discovery of cracks and to check out the effectiveness of the inspection instrumentation, methods, and techniques for application to the fleet (see 5.5.3). At the end of the fatigue/damage tolerance test program, a teardown inspection shall be performed.

5.3.4 Sonic fatigue tests. The contractor shall comply with the detail requirements for sonic fatigue tests as specified in MIL-A-8893. Prior to initiation of testing, the test plans, procedures, and schedules shall be approved by the Air Force. The sonic fatigue test program shall consist of a series

of laboratory and full-scale test programs to ensure that the aircraft basic design is structurally adequate for the acoustic loads which impinge on the aircraft. These loads result from power-plant noise during ground operation, pseudo noise in turbulent and separated airflow, and localized vibratory forces. In addition, consideration shall be given to significant combined environments, specially thermal effects.

5.3.4.1 Laboratory test program. As a part of the design-development and preproduction design-verification program (see 5.2.12), the contractor shall perform the necessary sonic fatigue test program on development components representative of the sonic critical structural areas utilizing the design acoustic environment from the sonic analysis.

5.3.4.2 Full-scale test program. As soon as a test aircraft is available, measurements shall be made of the acoustic environments on a full-scale aircraft to verify or modify the initial design acoustic loads/environment. The sonic fatigue-proof test shall be conducted on a representative aircraft (or its major components) to demonstrate structural adequacy for the acoustic service-life requirements of the aircraft. Proof/demonstration tests normally are accomplished by ground testing of the complete airplane with the power plants operating at full power for a time sufficient to assure design service life. However, use of major portions of the airplane in special nonreverberant ground test stands using the aircraft propulsion system as the noise source, or in high intensity noise facilities, may be acceptable. The ability to accelerate tests by compressing test time (by factors of 5 to 10 or more) in acoustic test facilities of adequate size and power offers great benefits in savings of time and costs. Qualified acoustic test facilities are available in the Air Force.

5.3.5 Flight and ground loads survey. The contractor shall comply with the detail requirements for the flight and ground loads survey specified in MIL-A-8871. Prior to initiation of testing, the test plans, procedures, and schedules shall be approved by the Air Force. The flight and ground loads survey program shall consist of operating an instrumented and calibrated airplane within and to the extremes of its limit structural design envelope to measure the resulting loads for the purpose of verifying the analytical loads and their distributions. Load measurements shall be made by the strain gage and pressure survey methods as agreed to between the contractor and the Air Force. The flight and ground loads survey shall include dynamic response and the thermal flight tests, as appropriate.

5.3.5.1 Objectives. The objectives of a loads survey shall be as follows: (a) determination and evaluation of loading conditions which produce the critical structural load and temperature distribution; (b) verification of the analytical structural loads and temperatures used to design the airplane structure; and (c) determination and definition of suspected new critical loading conditions indicated by previously conducted investigations of structural flight conditions and configurations within the design limit envelope.

5.3.5.2 Test airplane. The second airplane produced of each model shall be used to perform a flight loads survey and structural integrity flight demonstration. An additional airplane, sufficiently late in the production program to ensure obtaining the final configuration, shall be the backup airplane for structural flight tests and shall be instrumented as required during fabrication.

5.3.5.3 Dynamic response tests. The dynamic response flight tests shall be accomplished by measuring the structural loads and inputs while flying the airplane through atmospheric turbulence and during taxi, takeoff, towing, landing, refueling, and store ejection. The objectives shall be to obtain flight verification and evaluation of the elastic response characteristics of the structure to these dynamic-load inputs for use in substantiating or correcting the loads analysis, fatigue analysis, and for interpreting the operational loads data.

5.3.5.4 Thermal flight tests. Thermal flight tests shall be conducted as part of the flight-loads survey when the design performance requirements are such that structurally significant temperature conditions will be imposed on the airframe. The objective is to obtain flight measurements of the temperatures of various structural components for verification of the temperatures used in the design of the airframe.

5.3.6 Flutter tests. The contractor shall comply with the detail requirements for laboratory-type flutter tests as specified in MIL-A-8870. Prior to initiation of testing, the test plans, procedures and schedules shall be approved by the Air Force. The laboratory-type flutter tests shall consist of wind-tunnel flutter model tests, wind-tunnel aerodynamic model tests, ground vibration tests, influence coefficient and damaged and undamaged structural regidity tests, thermoelastic tests, limit load rigidity tests, and control surface free play and rigidity tests.

5.3.6.1 Aerodynamic model tests. Wind-tunnel aerodynamic model tests may be required for experimental determination of the nonsteady aerodynamic forces acting on the surface of the airplane. The objective is to improve the aerodynamic data which are used in the theoretical flutter analysis.

5.3.6.2 Ground vibration tests. The ground vibration tests shall consist of the experimental determination of the natural frequencies, mode shapes, and structural damping of the airplane or its components. The objective is to verify mass and stiffness characteristics which are used in the theoretical flutter analysis.

5.3.6.3 Structural rigidity tests. The influence coefficients and structural rigidity tests, thermoelastic tests, limit load rigidity test, and control surface free play and rigidity tests shall consist of the experimental determination of the structural elastic and free play properties of the aircraft and its components. The objective of these tests shall be to verify supporting data used in flutter analyses and flutter model design.

5.3.7 Flight flutter tests. The contractor shall comply with the detail requirements for flight flutter tests specified in MIL-A-8870. Prior to initiation of testing, the test plans, procedures, and schedules shall be approved by the Air Force. These tests shall be conducted to verify that the required damping in important modes of vibration exists and to substantiate freedom from flutter throughout the operational flight envelope.

5.3.8 Loads/environment spectra survey. The objective of the loads/environment spectra survey shall be to obtain a limited amount of service loads/environment data for assessing the applicability of the design loads/environment spectrum to initial actual service use experience. The flight data shall be obtained by the Air Force on a specified number of aircraft as part of the fleet management task described in 5.5. The number of aircraft to be used in the survey shall be specified in the request for proposal and the contract specifications. The contractor shall determine the instrumentation requirements and make the necessary instrumentation and data recording provisions in these aircraft as described in 5.4.4. The data acquisition shall start with the first operational aircraft. For the purpose of cost estimating and scheduling, it shall be assumed that duration of the program will be 3 years or when the total recorded operational flight hours equals one design lifetime whichever comes first.

5.3.8.1 Assessment of design loads/environment spectrum. The contractor shall use the reduced flight data as supplied from the Air Force aircraft structural integrity management information system (ASIMIS), or equivalent, to assess the applicability of the design (and fatigue test) loads/environment spectrum. In addition, the contractor shall update the design spectrum as necessary and, subject to Air Force approval, shall use the updated spectrum in the final analysis and strength summary tasks described in 5.4.1 and 5.4.2. It should be noted that this survey (and the resulting analyses) is a basic ASIP requirement but unlike the other full-scale tests described in 5.3, it is not for the purpose of proving compliance to the basic structural design requirements.

5.3.8.2 Loads/environment spectra survey requirements. The contractor shall utilize the flight loads/environment data obtained from a specified number of operational aircraft for a specified period of usage to assess the applicability of the design loads/environment spectrum to actual service usage. The scope (i.e. number of aircraft and hours of recorded data) of this survey shall be specified in the request for proposal and the contract specifications. The contractor shall update the design loads/environment spectrum as necessary and, subject to Air Force approval, shall use the updated spectrum in the final analyses, strength summary, and development of structural inspection and maintenance requirements for the fleet as specified in 5.4.

5.4 Fleet management data package (task IV). Tasks I through III cover the requirements that the contractor shall meet to provide airframe structures which have the required safety and durability throughout their design service life. However, this safety and durability will also be dependent upon the appropriate Air Force Commands performing specific inspection, maintenance, and possibly modification or replacement tasks at specific times throughout the service life (i.e., at specified depot level maintenance times and special inspection periods). To properly perform these tasks, the Air Force must have detailed knowledge of the task procedures and precautionary measures. In addition, in terms of the mission element mix, experience has shown that, due to a number of different reasons, the actual usage of military aircraft may differ significantly from the originally assumed design usage. It is also necessary that the Air Force have the specific technical tools and data to assess the effect of these potential usage changes on aircraft safety and on time to initial cracking. Task IV therefore describes the required elements of a fleet management data package which the contractor shall provide in a form that the Air Force can use to ensure safety throughout the operational life of the aircraft. As a minimum, this package shall contain the items specified in 5.4.1 through 5.4.5.

5.4.1 Final analyses. The contractor shall modify and revise all structural analyses of 5.2 to account for the significant results of all ground tests and flight surveys of 5.2.12 through 5.3.8. The final analyses shall be performed as early as possible consistent with the availability of ground test and flight survey results and shall be subject to Air Force approval.

5.4.2 Strength summary. This task is the summarization of the final analyses and other pertinent structures data into a format which provides rapid visibility of the important structures characteristics, limitations and capabilities in terms of operational parameters. The contractor shall summarize the final analyses and other pertinent structures data as specified herein and in the contract specification. The results of this task shall be approved by the Air Force.

5.4.2.1 Strength summary content. It is desirable that the summary be primarily in diagrammatic form (i.e., graphs, charts, sketches, envelope diagrams, et cetera) which should show the aircraft structural limitations and capabilities (e.g., strength, flutter, temperature, buffet, et cetera) as a function of the important operational parameters such as speed, acceleration, center-of-gravity location, and gross weight. It should also include brief descriptions of each of the major structural components (also preferably in diagrammatic form). These descriptions should indicate structural arrangements, materials, critical design conditions, fracture and fatigue critical areas, and minimum margins of safety. Appropriate references to design drawings, detailed analyses, test reports, and other detailed backup documentation shall be indicated. Additional detailed guidance as to the contents of the strength summary is contained in the strength summary section of MIL-A-8868.

5.4.3 Parametric analysis. The validity of the fatigue and damage tolerance analyses and the resulting life and inspection interval calculations are directly dependent upon the assumed mission element mix (i.e., the load/environment spectrum). From the results of in-flight service monitoring, the Air Force may find that the actual usage is different from that assumed and as a result that the baseline calculations will require updating. To allow the Air Force to readily adjust the basic life and inspection interval calculations with changes in mission element mix, the contractor shall provide crack-initiation and crack-growth calculations for selected critical areas of the structure in a generalized parametric form. The effort shall be coordinated with the Air Force prior to initiation to ensure that the results will be usable by the Air Force. The results of this analysis shall be subject to Air Force approval.

5.4.3.1 Compatibility with Air Force Analysis system. The ASIMIS can use the service monitoring input data described in 5.5 and the contractor supplied parametric fatigue analysis to determine the effect of mission element mix on predicted time-to-crack initiation. It is anticipated that in the future, the determination of inspection intervals based on crack-growth calculations can also be handled with this system. It is important that the contractor coordinate proposed parametric analysis efforts with the Air Force to ensure that the results will be compatible with the Air Force data processing and analysis system.

5.4.4 Instrumentation and data recording provisions. The contractor shall provide qualified functioning instrumentation and data-recording systems compatible with the ASIMIS in accordance with the requirements of this standard, the request for proposal, and the contract specifications. The contractor shall obtain Air Force approval of all instrumentation and recording equipment selections and the resulting provisions to be made in operational aircraft. Instrumentation and data-recording provisions shall be required to support the flight loads/environment spectra survey of 5.3.8 and to detect variations in mission usage on all other operational aircraft.

5.4.4.1 Flight loads/environment spectra survey provisions. The instrumentation and recording provisions for the aircraft used in the flight loads/environment spectra survey shall necessarily be more comprehensive than that required on other operational aircraft for routine mission element monitoring. It is anticipated that this instrumentation and recording equipment will consist of automatic data sensors (loads, strains, temperatures, pressures, linear and angular accelerations, et cetera), USAF standard multichannel recorders conforming to MIL-R-83165, converter multiplexers conforming to MIL-C-83166, and

manual data forms. The contractor shall select the specific instrumentation and recording equipment to best accomplish the survey task of 5.3.8, obtain Air Force approval of the selections, and make class A provisions for the selections in the specified operational aircraft.

5.4.4.2 Mission element monitoring provisions. The instrumentation and recording provisions for the purpose of routine mission element monitoring of all other operational aircraft shall be as simple as possible to minimize costs and the burden of data handling and processing. Simple counting accelerometers, VGH, and counting strain gage instrumentation shall be considered along with simplified manual data forms. The contractor shall select whichever is most suitable for the type of aircraft, shall obtain Air Force approval of the selections, and shall make the necessary class A provisions in the operational aircraft.

5.4.5 Service inspection and maintenance control. While the safety of past military aircraft has to a large extent been dependent upon in-service inspection, the importance of this inspection in a quantitative manner is recognized and inspection requirements herein (i.e., flaw size acceptance limits and inspection intervals) are made an integral part of the structural design process. Therefore, it is mandatory that the actual service inspection and maintenance procedures and frequency be consistent with that used in the design. Also, flaw size acceptance limits shall be consistent with service inspection flaw-detection capability as specified in MIL-A-8866.

5.4.5.1 Service inspection and maintenance control requirements. The conventional Air Force controls on service inspection, maintenance, and repair shall be through the use of technical orders and manuals. While these documents are normally broader in scope than structures and materials, per se, it shall be the contractor's responsibility to ensure that the structures and materials inputs on inspection and maintenance are complete and accurate. It is desirable that the contractor's structures and materials organization be responsible for these inputs and have final review and approval authority over the documentation prior to submittal for Air Force approval. In addition, design of field modifications and standard and special repairs shall be demonstrated by test and analyses to maintain all structural requirements of this standard and the contract specifications.

5.5 Fleet management (task V). Unlike ASIP task I through task IV described in the previous sections, task V will be primarily the responsibility of the Air Force and will be performed by the appropriate Air Force Commands with the minimum practical amount of contractor assistance. The objective will be that with task IV fleet management data package supplied by the contractor, the Air Force will be able to accomplish the following. Some specific details of these

task elements and orgainzational responsibilities are described in 5.5.1 through 5.5.4.

a. Support the loads/environment spectra survey described in 5.3.8

b. Monitor usage of operational aircraft including data processing and analysis

c. Perform the required structural inspections, maintenance, and repair throughout the life of the fleet

d. Maintain and evaluate structural performance records.

Specific delineation of contractor responsibilities in task V will be specified in the request for proposal and contract specifications.

5.5.1 Loads/environment spectra survey support. The Air Force will be responsible for the overall planning and management of the loads/environment spectra survey and will work with AFLC and the appropriate using command to: (a) establish data collection procedures and transmission channels within the Air Force, (b) train squadron, base, and depot level personnel as necessary to ensure the acquisition of acceptable quality data, (c) ensure the availability of adequate spare instrumentation and recording equipment, and (d) ensure that the results of ASIMIS are of acceptable quality and are obtained in a timely manner so that the contractor can complete the required spectrum assessment and analyses updating (reference 5.3.8 and 5.4.1).

5.5.2 Service monitoring program. Subsequent to the completion of the formal survey program described in 5.3.8, it is anticipated that the Air Force will elect to continue to operate either all or a portion of the instrumentation and recording equipment aboard the survey aircraft to support the service monitoring program. The service monitoring program will consist of tracking all operational aircraft throughout the life of the fleet to determine actual service usage (i.e., mission element mix) and the potential impact of this usage on estimated crack-initiation times, crack-growth rates, and inspection and maintenance requirements. As specified in 5.4.4, the minimum practical amount of instrumentation will be used for this purpose. The ASIMIS will be used to process and analyze the recorded data. As noted in 5.4.3, the parametric analysis supplied by the contractor is an essential element in this effort. The initial planning, development, and management of the service monitoring program and the required computer applications program will be the responsibility of the Air Force with the actual data collection and reduction being performed by the using command and AFLC. Subsequent to transferring the aircraft program from AFSC to AFLC, AFLC will accept responsibility for the total program.

5.5.2.1 Assessment of service usage. In addition to data collection, data transmission, spares availability, training, and data processing activities similar to that described in 5.5.1, the total effort will include the engineering assessment of the ASIMIS results. The effect of actual service usage on

planned inspection intervals and inspection and maintenance procedures will be determined and, as necessary, adjustments will be made to ensure continued safe operation.

5.5.3 Service inspection, maintenance, and repair. The appropriate Air Force Command shall conduct structural service inspections, maintenance, and repair in accordance with the detailed requirements and procedures developed by the contractor and specified in Air Force technical orders and manuals (reference 5.4.5). AFLC, in conjunction with the using commands, will modify the technical orders and manuals to adjust the frequency and content of the inspection and maintenance requirements as found necessary due to the results of the service monitoring program (reference 5.5.2) and servicing inspection experience.

5.5.3.1 Inspection details. The type, extent, and frequency of structural service inspection and maintenance planned for a specific aircraft system will vary as a function of the original inspection interval requirements specified in the contract specifications, the specific structural design approach (or approaches) selected by the contractor to comply with the service life design requirements (reference 5.1.2.1 and MIL-A-8866), and any changes incorporated due to the results of ground tests and flight surveys (reference 5.2 and 5.3). As a minimum the service inspection will include periodic field inspections and depot level (IRAN) inspections. The criteria for IRAN are established in T.O. 00-25-4. Additional special scheduled inspections may be required if the design structural inspection interval (as specified in the contract specifications or as subsequently modified as a result of test or service data) and the IRAN interval are incompatible. Analytical condition inspections or complete teardown inspections are also sometimes performed on one or more operational aircraft after a period of service use. The decision as to whether or not such an inspection will be made depends upon the specific characteristics of the aircraft system and the accumulated service experience.

5.5.4 Structural performance records. Records of fleet inspection, maintenance, and repair and the output from the service monitoring program provides a large body of statistical data which continually grows with time. The usefulness of this data for assessing fleet safety and durability also grows with time. AFSC/AFLC, as appropriate, will be responsible for maintaining these records, for performing periodic evaluations and for making recommendations with regard to potential modifications, improvements, component replacements, and restrictions on service usage. This standard and the supporting MIL-A-8800 series structural specifications will be used as a guide for the design development, and verification of any structural modifications or improvements resulting from these recommendations.

6. NOTES

6.1 Data requirements. The data requirements in support of this standard will be selected from the DOD Authorized Data List (TD-3) and will be reflected in a contractor data requirements list (DD Form 1423) attached to the request for proposal, invitation for bids, or the contract as appropriate.

6.2 Relationship to systems engineering management. The conduct of the work efforts by the contractor in achieving airplane structural integrity is to be in accordance with the System Engineering Management Plan for that particular airplane system and in accordance with the system safety plan (MIL-STD-882).

6.3 Tasks. The tasks (see 4.2) specified herein are referred to as "Phases" in AFR 80-13 and a correction will be made during the next revision to that regulation. The change is necessary to avoid misrepresentation of the ASIP functional tasks with the time-oriented system life-cycle phases.

Custodian:
Air Force - 11

Reviewer Activities
Air Force - 01, 10, 16

Preparing Activity
Air Force - 11

Project No. 1500-F010

SUPPLEMENT (G)

ADVANCED AIR SUPERIORITY FIGHTER WING STRUCTURES PROGRAM - FOLLOW-ON PROGRAM PLAN

FZM 6134
1 May 1973

ADVANCED AIR SUPERIORITY FIGHTER

WING STRUCTURES PROGRAM

FOLLOW-ON PROGRAM PLAN

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FOREWORD

To ensure continued air superiority, future weapon systems must reliably achieve maximum performance at minimum costs. The purpose of this program is to provide design methodology and advanced structural configurations which will satisfy the requirements of future aircraft. To accomplish program objectives, a multi-phase program is required. The preliminary portion, designated Phase IA, is currently being pursued by the General Dynamics Operation and by the Northrop Corporation in parallel programs.

The wing box of the F-111F aircraft was chosen as the structural component for the Fort Worth Operation Program. Structural integrity of the basic F-111 wing has been verified by a comprehensive program of testing and analyses. Existence of this data, and the test facilities used to generate it, will provide a sound basis for evaluating advancement of airframe technology.

The current contract, which covers Phase IA only, provides for preliminary design and analyses and limited material screening tests. The output of this initial phase of the program will be the identification of three weight and/or cost saving configurations of the F-111 wing, plus the design methodology mentioned above.

The follow-on effort will complete the design and evaluation of configurations currently being generated. A single configuration which best meets the program objectives of maximum performance at minimum costs will be included in two full scale wing boxes. These boxes will undergo static and fatigue testing, with selected damage tolerance testing presented as an additional option. The results of this testing will be compared to the basic F-111 test results to assess the degree of technology advancement which the design methodology provided.

This input to the Air Force is intended to define and schedule the follow-on tasks and provide an estimate of the resources (time, material, and money) necessary to complete the follow-on effort.

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1.0 INTRODUCTION

Recent structural difficulties in Air Force Weapon Systems have focused attention on problems which heretofore concerned only the structure and materials specialists. These difficulties served to emphasize the need for a long term Advanced Development Program (ADP) concerned with improvements in metallic aircraft structures. Major hardware programs, which integrate and exploit new design concepts, fracture mechanics, analysis methods, design criteria, materials, manufacturing methods, non-destructive inspection and information transfer methods are required as part of this ADP to anticipate and solve critical aircraft structural problems prior to the acquisition of new Air Force Weapon Systems.

1.1 Over-all Program Goal - The over-all goal of the ADP is to reduce the risk of structural failure of future Air Force aircraft by increasing their structural reliability, integrity, and efficiency. This goal shall be achieved through the integration, exploitation, application and evaluation of new or improved structures, materials and manufacturing technologies. Development and application of fracture mechanics techniques are integral to this program. The increased structural reliability, integrity and efficiency of the assemblies built and tested under this ADP shall result in a payoff to the Air Force by a thorough, timely information transfer of the improvements in the structures, materials, and manufacturing technology base to future systems.

1.1.1 Approach to Achieve Goal - In the approach to achieve this goal, it is anticipated that iterative design and test effort beyond that usually occurring during a system development will be required. Moreover, significant additional iterations through the major phases of the design and fabrication planning process are likely. In addition, results of this program may indicate a high payoff to be obtained by repeating all or a portion of this program. Consequently, the necessity for iterations at all levels and during all phases of the program should be accounted for in the contractor's selected approach. The contractor shall maintain complete records of the program efforts to insure effective and productive use of iterative design.

These records will be especially useful during future programs to identify those iterative loops which produce the most significant results.

1.2 Specific Objectives - The objective of this follow-on program plan is the design, fabrication and testing of a full scale wing box structure for an advanced air superiority fighter and the timely dissemination of the resulting body of knowledge and experience. The structure shall have Air Force approved potential for weight and/or cost improvement over the selected baseline wing structure through the application of new and emerging technology.

This advanced assembly shall be designed to comply with the structural design criteria of the baseline. In addition, the fatigue and damage tolerance requirements of MIL-STD-1530 and MIL-A-008866A (including proposed Revision D, dated 18 August 1972) shall be applicable to both the advanced assembly and the baseline.

2.0 SCOPE

The follow-on program is structured to evaluate the potential of the three design configurations selected from the comprehensive analytical studies of Phase IA to provide a reliable, advanced fighter wing that will achieve maximum payoff. A five phase program is planned for this effort as follows:

<u>Phase Ib - Preliminary Design and Analysis</u>

Preliminary design and analysis of three configurations selected from the current Phase IA program will be performed during this phase. Trade studies, material testing, and design verification testing will also be accomplished. Parameters for manufacturing processes and NDI methods will be developed for use later in the program. After careful evaluation, two configurations will be chosen for the next phase of the program.

<u>Phase II - Detail Design and Analysis</u>

Detail design and analysis of these two configurations will be performed during this phase. Both configurations will undergo several iterations of study

and analysis to arrive at an optimum design. Additional material testing and pre-production validation testing of selected components will also be accomplished in this phase. A final design will be chosen for fabrication and testing.

Phase III - Fabrication

Two identical left hand full scale wing box structural articles will be built to the engineering drawings developed in Phase II. Test plans, including instrumentation requirements, will be finalized. Cost and weight records will be maintained during fabrication to verify earlier estimates.

Phase IV - Test and Evaluation

A static test, fatigue test, and optional damage tolerance verification testing will be accomplished on the assemblies fabricated in Phase III. The testing will be compatible with the test program already conducted on the basic F-111 wing.

Phase V - Information Transfer

The purpose of this portion of the program is to insure proper documentation of program results and the reporting of significant accomplishments in such a manner that they are useful to future Air Force systems programs. Documentation will include monthly progress reports, semi-annual technical reports, phase reports, test reports and a final report. Movies, slides, and viewgraphs will be used to ensure clear concise reporting.

This plan includes those items which General Dynamics Corporation feels are necessary to evaluate the design methodology.

The schedule for accomplishing this program is shown in Figure 2-1.

ADVANCED AIR SUPERIORITY FIGHTER WING STRUCTURE PROGRAM SCHEDULE

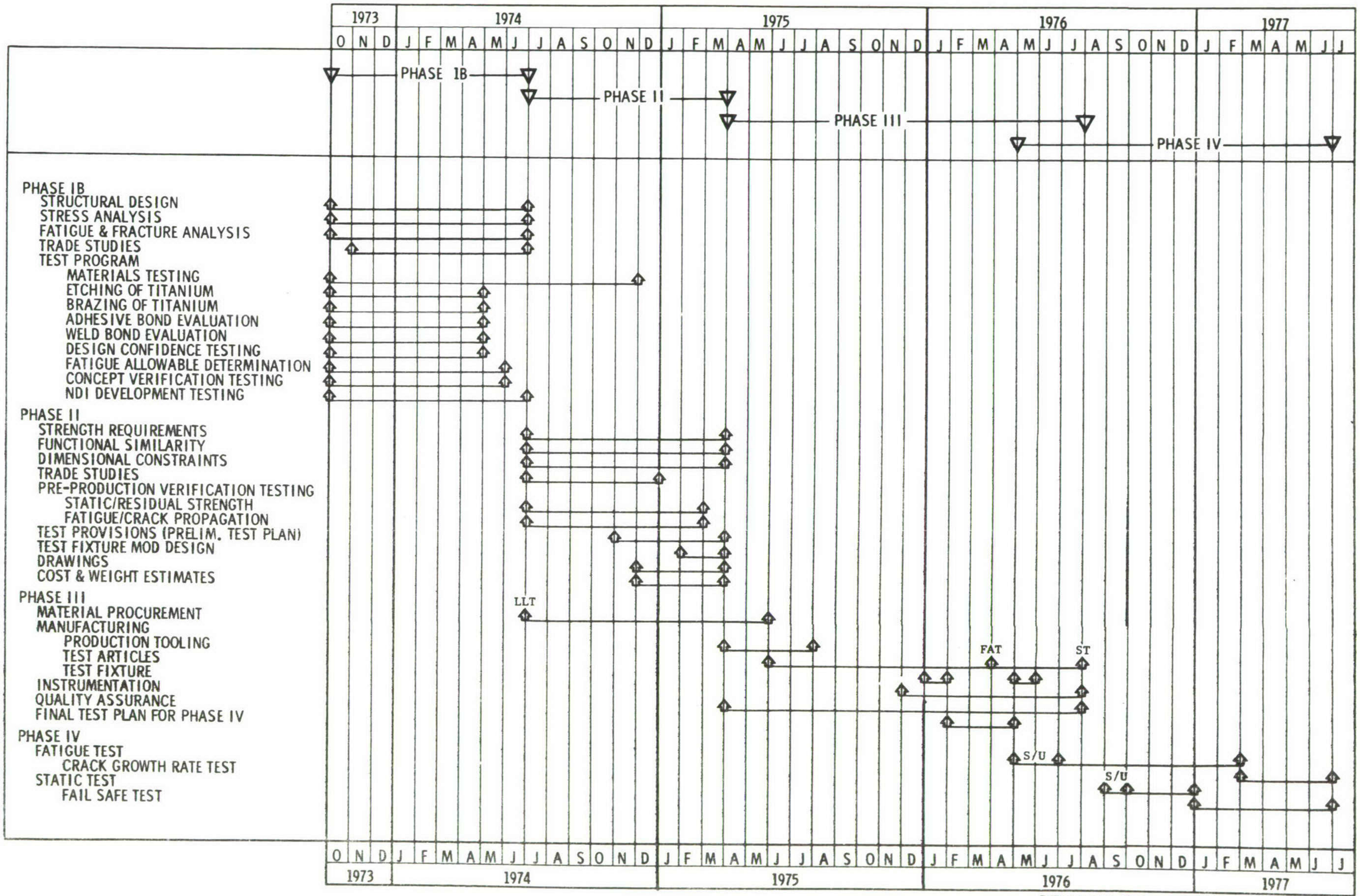


FIGURE 2-1

ADVANCED AIR SUPERIORITY FIGHTER WING STRUCTURE PROGRAM SCHEDULE

PHASE IV - SINGLE WING TESTING

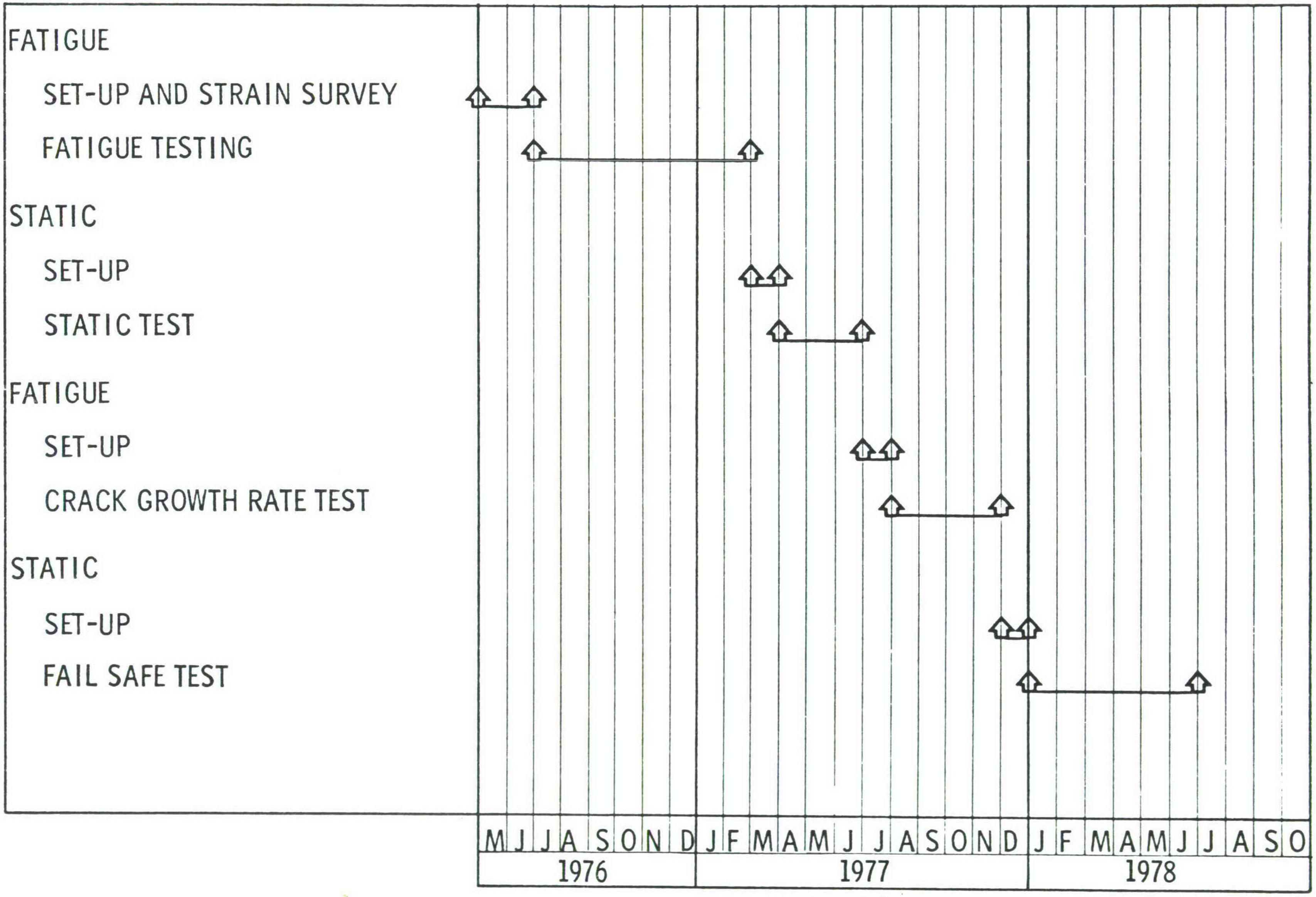


FIGURE 2 - 2

INDUSTRIAL ENGINEERING & SCHEDULING
SUPPLEMENT #1 TO SCHEDULE NO. 40-0-2"A"
17 MAY 1973

3.0 PROGRAM DISCUSSION

Details of the program tasks are shown schematically in Figure 3-1 and are discussed in this section. The following elements are contained in the plan:

1. Material testing will be conducted to establish a statistically significant design allowables data base.

2. Design iteration will be accomplished to optimize the structure using the statistical allowables data base.

3. Analysis techniques will be used to demonstrate the compliance of the structure with static, fatigue and damage tolerance criteria.

4. Detail design drawings will be prepared to allow production of hardware.

5. Manufacturing and Quality Assurance plans will be formulated to establish fabrication and inspection criteria.

6. Design development and validation tests will be planned to generate design information and to demonstrate the feasibility of the configurations selected in Phase IA.

7. Design verification tests of a complete wing box will provide proof of compliance in meeting the static, fatigue, and damage tolerance criteria.

8. Information transfer will ensure dissemination of all advanced technology developed during the program.

3.0.1 Planning Requirements - As a part of this program, the contractor will prepare a Master Planning Document detailing the planned use of resources to execute this program. The Master Planning Document will be revised to reflect significant changes as they occur and to provide increased detail as portions of the program become better

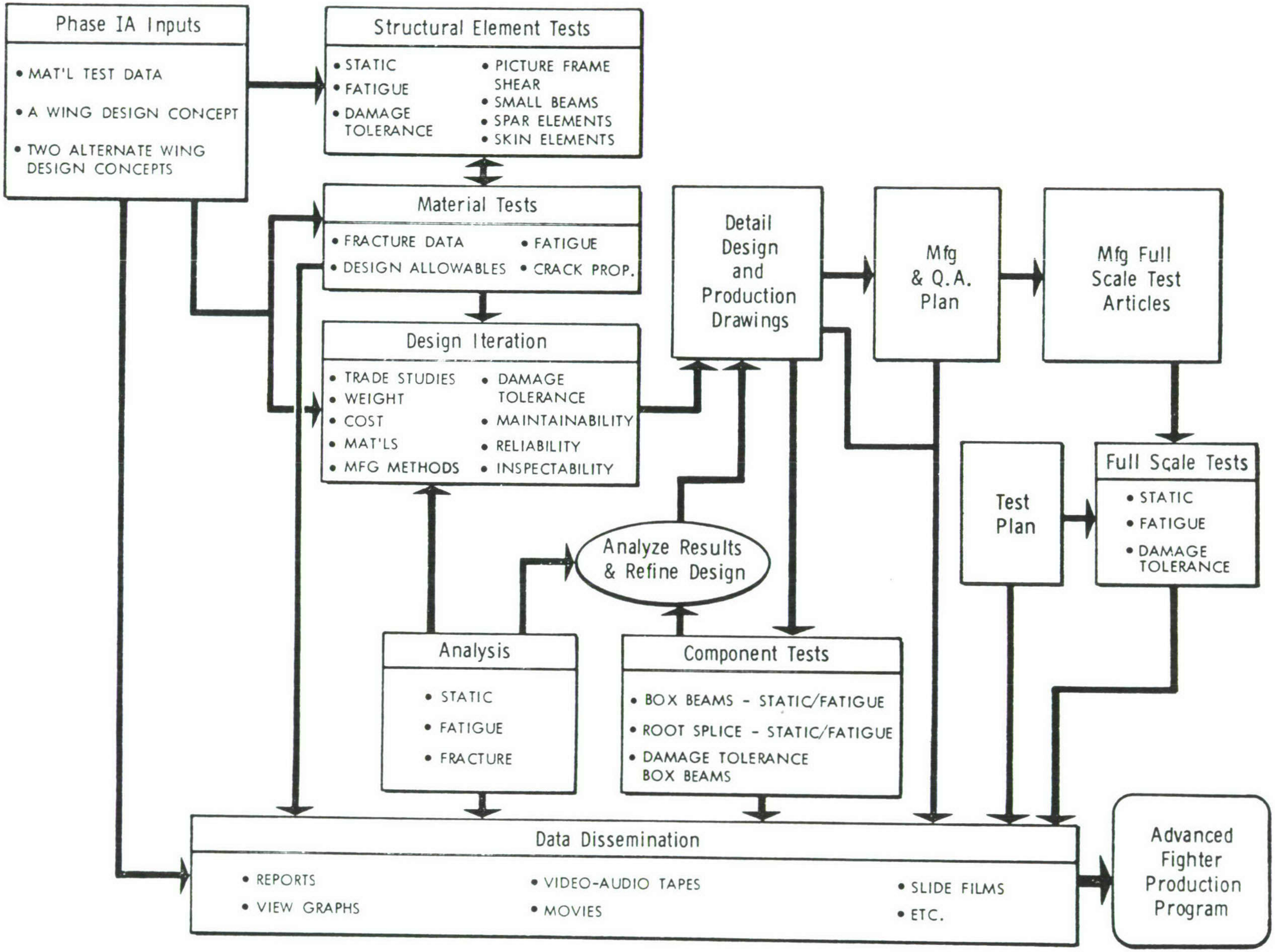


Figure 3-1 Follow-On Program Study Plan

defined. These revisions will be evolved in discussion with cognizant Air Force Management and Technical personnel Revisions are subject to written approval of the Air Force Contracting Officer. All expenditures of resources must be in accordance with the latest approved Master Planning Document.

3.0.2 Planning Topics - The topics of the major section of the Master Planning Document shall be Preliminary Design, Detail Design, Fabrication, Test, and Information Transfer. The test plans required by paragraphs 3.1.6, 3.2.8, and 3.3.8 will be submitted as revisions to the Master Planning Document.

3.0.3 Master Planning Document Coverage - At the request of the Air Force Contracting Officer, additional coverage will be provided by the contractor in the Master Planning Document.

3.1 Phase Ib - Preliminary Design and Analysis

Preliminary design and analysis will be performed on each of three configurations of wing box structures selected from Phase Ia of the program. The effort will proceed in accordance with the Master Planning Document, and will include trade studies and testing to support the design and analysis.

3.1.1 Structural Design - The configuration selected for the follow-on program will undergo several iterations of study and analysis to arrive at an optimum design. Each iteration will include arrangement optimization, finite element analysis, damage tolerance assessment, NDI evaluation, producibility, cost estimates, and a reevaluation of specific areas that require damage tolerance considerations (i.e., areas that cannot be inspected or where stress levels are too high for safe-life design).

As the designs are developed, they will be monitored, reviewed, and evaluated for cost and weight optimization. Surveillance of the structural arrangement, environment, material selection, construction, number of parts, and methods of manufacture will be maintained from contractual authorization until final release. Design concepts and drawings will be monitored and alternate design approaches will be considered, as necessary, to assure structural

efficiency; management will be advised of potential cost and weight changes; and a follow-up of management design decisions will be made to assure a minimum cost/weight design.

The cost impact of various details and manufacturing processes will be developed for use in the evaluation to select the final design. Consideration will be given to cost/worth ratios of testing, NDI, and quality assurance of the designs and the manufacturing processes involved.

3.1.2 Stress Analysis - Stress analysis during the follow-on program will be accomplished as a systematic review, optimization and certification of the items of structure throughout all iterative procedures. Generally this will be the same procedure used in the early portion of the program, but it will be expanded or improved as needed to perform the following tasks.

Preliminary designs selected at the end of Phase IA will be reanalyzed in the light of materials allowables from continued coupon testing. This is expected, in several iterations, to lead to advantageous changes in the placement of materials. A system for recording the results of this iterative procedure will be developed so that these results are readily accessible for review.

The results from the subcomponent tests will be reviewed and applied to validate or modify the analytical procedures employed up to this point in the program. It is quite possible that reiteration will result in improvements in the design.

The detail design of a major candidate and its alternates will be monitored continuously for structural adequacy. Stress inputs will be made regularly for purposes of keeping current the weight-cost-life studies. The analysis methods described in Phase IA will be used with whatever updating is made in light of tests and experience in this program.

The analysis experience at this point will be utilized for indications of differences between analysis and test results. Where differences exist, efforts will be made to reconcile these differences.

Final results of the program will be supported by the necessary analysis to validate the choices and trade-offs

made in the program. Analysis reports will be submitted in formats that have been approved for their clarity of presentation and completeness of coverage.

3.1.3 Fatigue and Fracture Analysis - To ensure that each design meets the new criteria specified in MIL-STD-1530, MIL-A-8866, and MIL-A-8867, a complete fatigue and fracture analysis must be performed. Preliminary analyses will be made during Phase Ib and will be refined in Phase II. The basic approach is defined below.

3.1.3.1 Fatigue Analysis - Fatigue analysis based on Miner's theory of cumulative damage will be used to verify adherence to the fatigue requirements. Fatigue control points will be defined for the principal tensile loaded elements. These control points will be selected on the basis of stress analysis results and consideration of design detail. Control point limit stresses (2/3 ultimate stress) will be determined for the selected baseline design conditions from the stress analysis results and used to compute a fatigue stress spectrum for each control point. The stress concentration factor, K_T, for each control point will be estimated on the basis of fine-grid stress analysis and/or K_T values based on prior fatigue test experience. Given the stress spectrum and the K_T level, the fatigue damage for the particular control point will be calculated as a function of stress level using computer procedure AOR and the baseline flight-by-flight fatigue spectrum. The ultimate stress level for which the fatigue damage is 1 for a scatter factor of 4 is the allowable stress. The maximum permissible allowable stress is the ultimate tensile strength of the material.

3.1.3.2 Reliability Analysis - Reliability analyses will be updated to reflect the additional data generated in the follow-on program. Analysis techniques will be identical to those described in Phase IA. More adequate statistical data on failures, their causes, and subsequent repairs will be available for establishing the relationships between weight, cost, and reliability for the selected configurations. The additional material allowables data base will be used to update the material characterizations.

3.1.3.3 Fail-Safe Analysis - The same finite-element math-models used for stress analysis will be used to conduct a residual strength analysis of the multi-element fail-safe designs. In the complete finite element simulations, individual elements can be reduced in size or eliminated to simulate failure. Orthotropic elements are used to simulate inability to react shear and normal forces along a crack or line of separation. Stress distributions in the altered structure are plotted and tabulated using the stress-analysis output format.

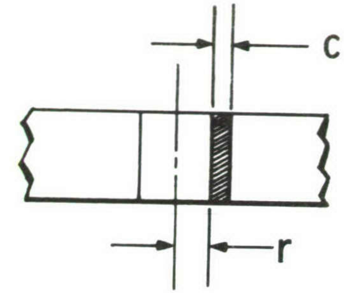
3.1.3.4 Crack Growth Analysis - A crack growth analysis will be conducted to determine the safe crack growth characteristics. An initial crack of the size specified consistent with fracture criteria is assumed to exist in the most unfavorable orientation with respect to the applied stress and the material properties. The growth of this flaw in the anticipated chemical, thermal and cyclic-stress environment will be computed using constant amplitude crack growth data an an analysis model that satisfactorily accounts for load interaction effects due to variable-amplitude fatigue cycling.

Control points will be defined for the primary tensile-loaded fracture critical elements of each candidate design. These control points will be selected on the basis of the finite element stress analysis results and a consideration of the design detail. For each control point, the functional relationship between K and crack length will be defined using existing stress intensity models coupled with estimating techniques or from the results of finite element fracture analysis. Typical existing models are summarized in Figure 3-2. Experimental data that relate the crack growth rate and critical crack size to the applied stress intensity level will be generated as is discussed in the Material Test Plan (reference paragraph 3.1.6). Crack growth life will be calculated using the baseline service loads flight-by-flight fatigue spectrum by integrating the growth rate between the limits set by the assumed initial flaw size and the final size based on critical fracture toughness and residual strength requirements. The Wheeler crack growth model will be used to account for load sequencing and environmental effects, Figure 3-3.

• BOLT HOLES (Bowie Model)

$$K = \sigma \sqrt{\pi c} \quad F(c/r)$$

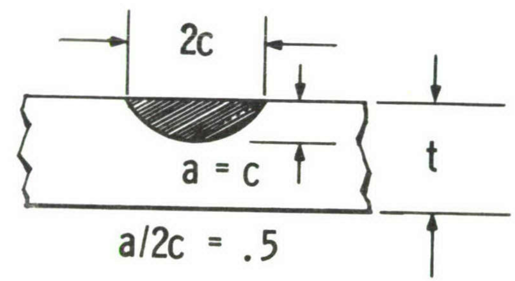
$$C = \frac{K^2}{\pi \sigma^2 [F(c/r)]^2}$$



• SURFACE FLAW (Part through)

$$K = M_K \frac{1.1 \sigma \sqrt{\pi a}}{\sqrt{\phi^2 - .212 (\sigma/\sigma_y)^2}}$$

$$a = \frac{K^2 [2.46 - .212 (\sigma/\sigma_y)^2]}{1.21 \pi \sigma^2 M_K^2}$$



• SURFACE FLAW (through the Thickness)

$$K = \sigma \sqrt{w \tan \left(\frac{\pi a}{w} + \frac{K^2}{2w \sigma_y^2} \right)}$$

DERIVED (FOR $w \geq 6''$)

$$2c = \frac{1}{\pi} \left(2.0 - \frac{\sigma^2}{\sigma_y^2}\right) \left(\frac{K}{\sigma}\right)^2$$

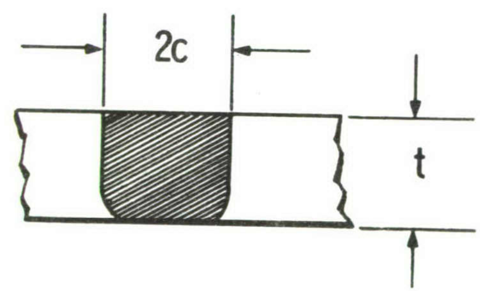
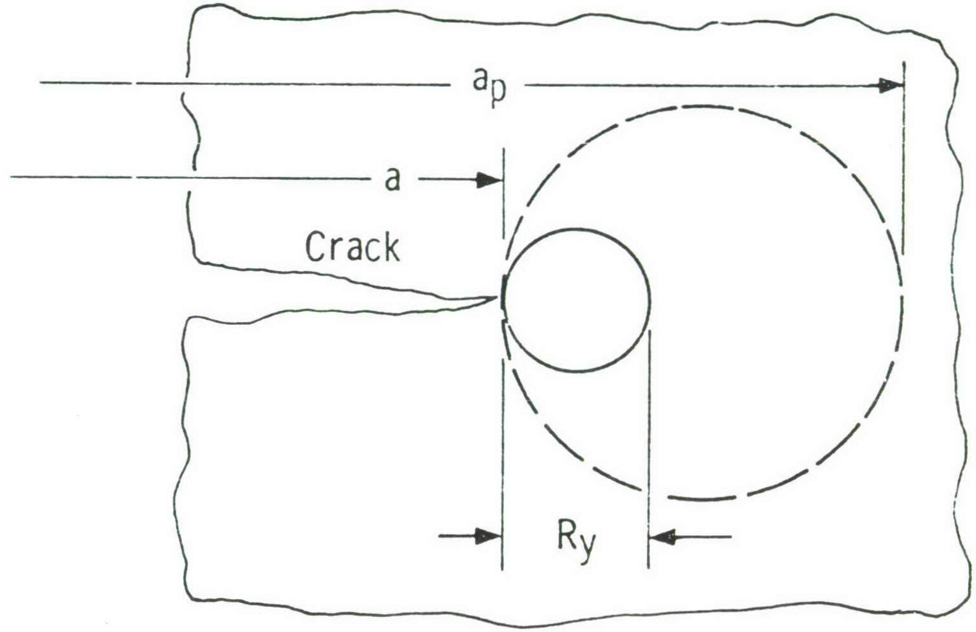


FIGURE 3-2 Stress Intensity Models

- $a_n = a_i + \sum_{i=1}^{n} C_{pi} \left[da/dN = f(\Delta K_i) \right]$
- $a + R_y \geq a_p$

  $C_{pi} = 1$
- $a + R_y < a_p$

  $C_{pi} = \left(\frac{R_y}{a_p - a} \right)^m$
- $\frac{da}{dN} = f(\Delta K)$



Where,

R_y = Current Yield Zone Size

$(a_p - a)$ = Distance from Crack Tip to Elastic-Plastic Interface

m = Shaping Exponent Established Experimentally

FIGURE 3-3 Wheeler Retardation Model

The value of the retardation exponent, m, required for the Wheeler model will be determined empirically in the spectrum fatigue test program. Thermal effects on crack growth rates will be neglected; however, the critical crack size will be based on the fracture toughness at -65°F, the minimum design temperature. Sustained load crack growth, da/dt, will be assumed negligible providing stress intensity computed using the maximum 1g stress level and the instantaneous crack length is less than the stress corrosion threshold, K_{ISCC} (material selection should be such that this is the general case).

The allowable ultimate stress for control points in fracture critical parts will be determined on the basis of crack growth analysis. Curves will be prepared by plotting the allowable ultimate stress as a function of the assumed initial flaw size and the specified inspection interval. The specific methodology for generating stress allowables curves, shown in Figure 3-4, is as follows:

1. Calculate a series of crack growth curves (crack length vs. number of flights) using a series of factors on stress level

2. From (1) dtermine the maximum initial crack size that permits one inspection interval of subsequent growth as a function of the maximum ultimate stress in the spectrum

3. Plot the allowable ultimate stress as a function of initial flaw size for typical periods of unrepaired service usage of ½, 1, and 2 lifetimes.

4. Determine the allowable ultimate stress level in accordance with the initial flaw size and inspection interval requirements.

Critical crack size will be calculated for each control point using the residual strength specified for flawed structure in the damage tolerance criteria. For each control point, the stress state existing at the onset of fracture will be defined as follows:

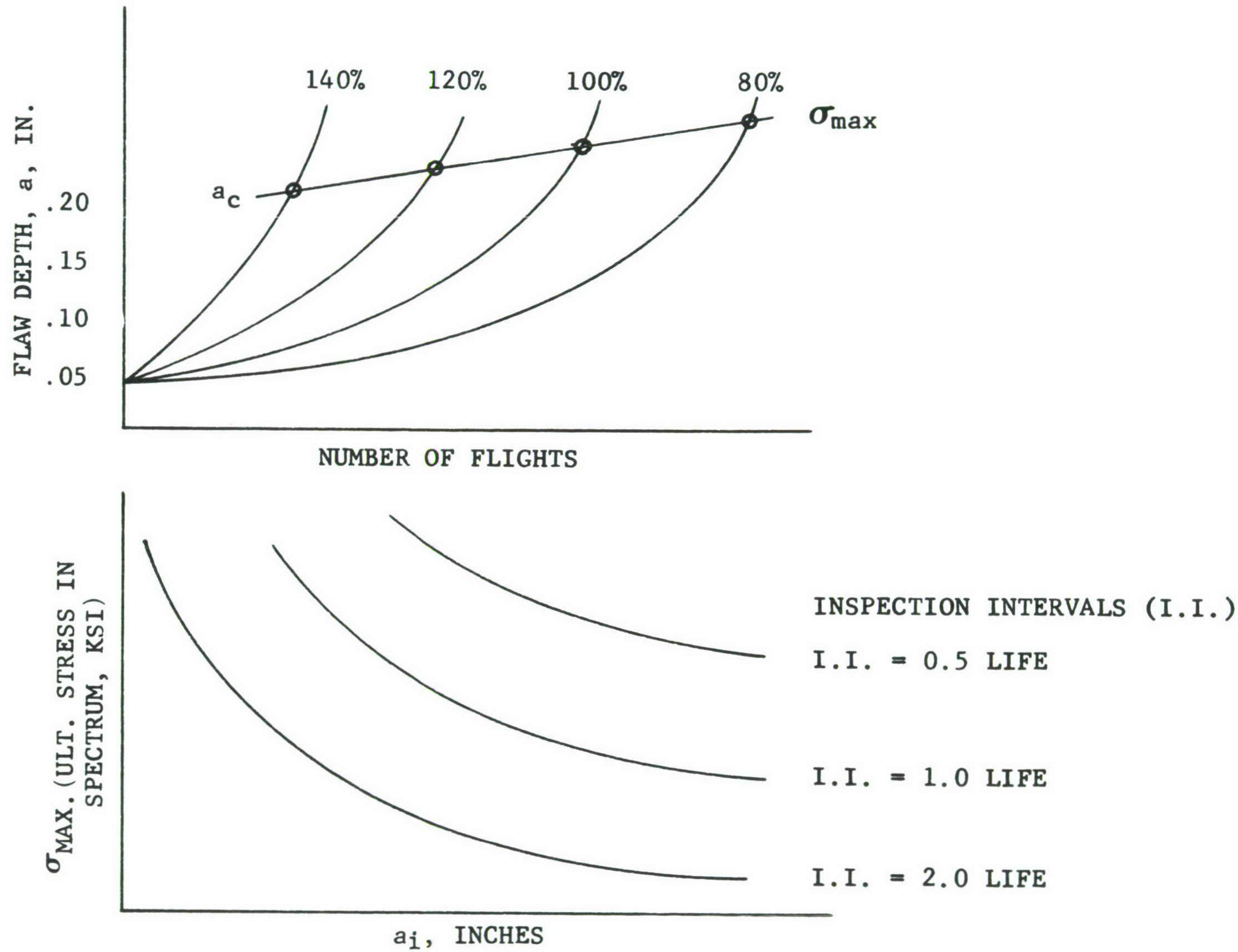


FIGURE 3-4 Crack Growth Allowables

Plane Stress $t < \frac{1}{2.5}\left(\frac{K_{IC}}{\sigma_{YS}}\right)^2$ where t = part thickness

Mixed Mode $\frac{1}{2.5}\left(\frac{K_{IC}}{\sigma_{YS}}\right)^2 < t < \left(\frac{K_{IC}}{\sigma_{YS}}\right)^2$

Plane Strain $t > \left(\frac{K_{IC}}{\sigma_{YS}}\right)^2$

Plane strain fracture toughness, K_{IC}, will be used for critical crack length calculations in control points classified as plane strain. The fracture toughness for mixed mode and plane stress will be based on the K_C vs t curve generated as part of the test program (reference paragraph 3.1.6).

3.1.3.5 Risk Assessment Analysis - The evaluation of structural in-flight risk is accomplished using statistical and probability techniques. Specifically the risk assessment analysis can be used to:

- evaluate individual aircraft of fleet structural probability of survival during service life considering initial NDI/proof test prior to operational usage and considering subsequent inspections during service life
- establish which of the individual structural parts are most critical
- investigate the sensitivity of probability of survival values to variations in parameters such as usage, initial flaw size distribution etc.

The major evaluation tool used in this analysis is the computerized risk assessment model. The risk assessment model is basically a set of mathematical and probability equations which describe a close approximation of the probability of structural survival during aircraft operations in the service environment. The equations are a function of those parameters that influence failure including:

- Initial flaw size distribution within each part which includes the influence of the non-destructive inspection (NDI) probability of flaw detection and of proof test maximum flaw length.

- In-flight flaw growth predictions which reflect service environment.

- Time to failure distribution for a part with an initial flaw of given size which includes the dispersion of part times-to-failure due to variations in load history, crack growth, K_{IC}, etc.

- Periodic inspections accomplished at specific time intervals to insure the integrity of primary structure.

The initial flaw size distribution, the flaw growth predictions, the failure distribution and the periodic inspection information serve as input data into the risk assessment model. The model translates these inputs to probability of survival values for a single aircraft and risk associated with the critical structure in the crack propagation failure mode.

Flaw sizes and crack growth curves for use in risk assessment of the final design must be provided. These curves will be supplied from the final crack growth analysis effort.

3.1.4 Trade Studies - Trade Studies will be conducted where appropriate to determine the impact of various materials and material properties, structural shapes and arrangements, methods of analysis, fabrication techniques and evaluation procedures on the three (3) configurations. As a minimum, the impact on weight, cost, strength, fatigue life, inspectability, environmental sensitivity, and relative improvement of the combined structures, materials, manufacturing technology base will be assessed. Recommendation of the configurations to be continued into detailed design will be made on the basis of a merit rating system shown in Table I.

Results of all trade studies will be documented and reported in the Phase Ib Final Technical Report.

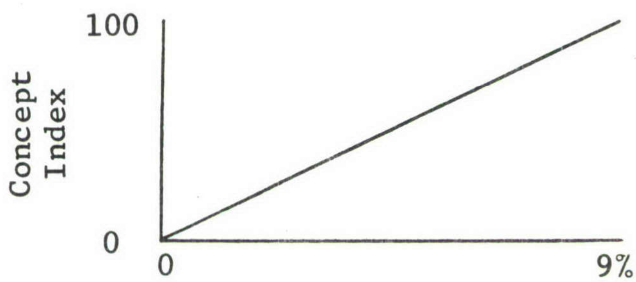
Repeated usage of the rating system referenced above has suggested some potential improvements in evaluating some of the elements of the system. During the early months of the Follow-On Program, these suggestions will be evaluated. Some of the suggestions are presented below:

Technology Advancement (30%)

o Design Concepts - (9%) - In Phase Ia, scores for this category were calculated based on the number of innovative features. A possible improvement would be to further rate each innovation as to its application. For instance, innovations which make maximum use of new materials would be rated between 0 and 40; innovations which advances manufacturing process would be rated between 0 and 40; innovations which are structurally unique would be rated between 0 and 20. These ratings would be totaled and a value up to 9% would be read from a curve for this category. An example is discussed below.

	Index
* -Makes maximum use of new and limited usage materials consistent with overall design objectives and criteria.	0 to 40
* -Demands improvement and/or development of manufacturing processes to achieve design objectives.	0 to 40
* -Possesses unique structural elements and arrangements which permit better compliance with design criteria and/or objectives	0 to 20
TOTAL CONCEPT INDEX	0 to 100

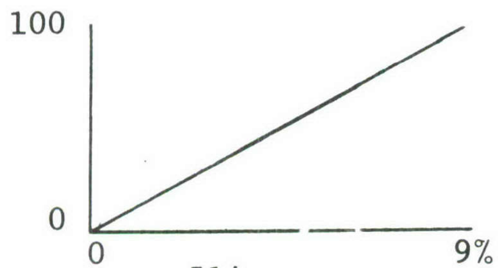
* Consider degree of development risk within program constraints



o Manufacturing (9%) - A similar procedure can be used to evaluate the Manufacturing Aspects of Technology Advancement. For instance, processes which reduces cost may be rated between 0 and 15, processes which increase structural reliability may be rated between 0 and 40, a process which improves producibility may be rated between 0 and 15, and a process which improves structural efficiency may be rated between 0 and 30. Totalling these ratings for a concept will allow a value up to 9% to be read from the curve. An example of this is shown below.

Manufacturing process development which has potential widespread application and is either an entirely new process or extension/modification of an existing process/processes - limited or no service experience-

<u>Mfg Index</u>	<u>Index</u>
1. Process results in potential cost decrease over existing methods	0 - 15
2. Process increases struct integ and reliability by reducing struct. complexity, inc. Q.C. Possible, improves inspection, etc.	0 - 40
3. Process improves producibility	0 - 15
4. Process permits improved structural efficiency (Dec. Wt. due to Tol. control, eliminates joints & splices, no reduction in parent metal strength as a result of joining such as welds, etc.	0 - 30
TOTAL CONCEPT INDEX	0 - 100



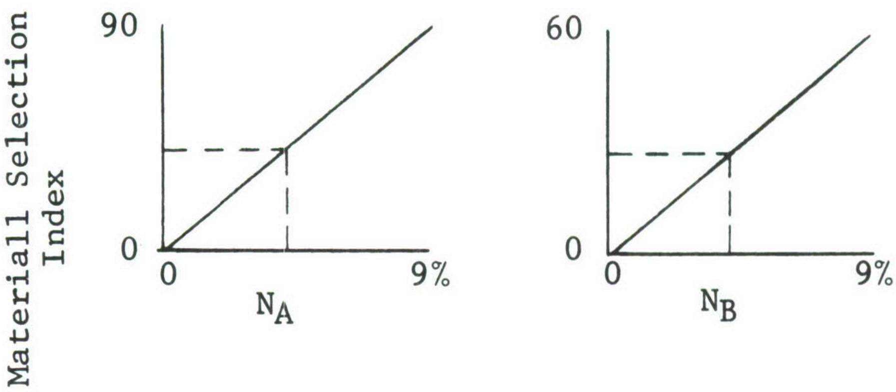
o Materials (9%) - Improvements in this category are based on the fact that material selection for the design of most air vehicle structures depends on how closely its physical and mechanical characteristics match the design requirements. Generally, the characteristics listed in the suggested rating criteria are those which most influence the selection. If, for example, two selection criteria are established as follows:

"A" selection criteria is intended for use with primary tension applications while "B" selection criteria is for use with stability critical structures, shear webs or other non-fracture sensitive structures.

Using this as a basis, a set of material characteristics can be established. A sample is shown below:

Material Characteristic	Selection Index	
	Application "A"	Application "B"
Structural Efficiency Index $\frac{Y}{}$	0 - 20	0 - 5
Cost (Raw + Fabrication)	0 - 20	0 - 20
Fracture Toughness - K_{IC}	0 - 10	0 - 5
Stress Corrosion Susceptibility K_{ISCL}	0 - 15	0 - 10
Availability	0 - 5	0 - 5
Corrosion Characteristics	0 - 10	0 - 10
Fatigue (Notch Sensitivity)	0 - 10	0 - 5
Maximum Possible Index	90	60

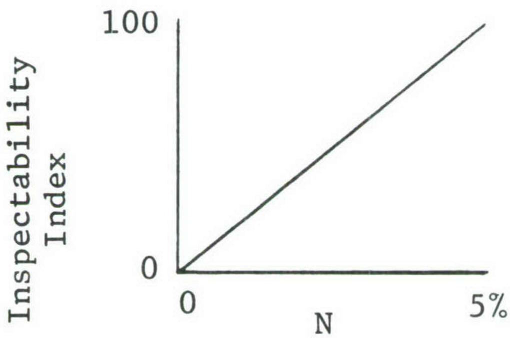
Using this material selection index, a rating can be made for Type "A" design (N_A) or Type "B" design (N_B). The best concept would receive the maximum rating of 9%. This is demonstrated below



Abilities (10%)

Similar techniques can be used to rate the "Abilities." Samples for all five categories are presented below. Each technique is based on defining index values for particular desireable features and computing a total Index number. This index number is then used to read a rating number from a graph.

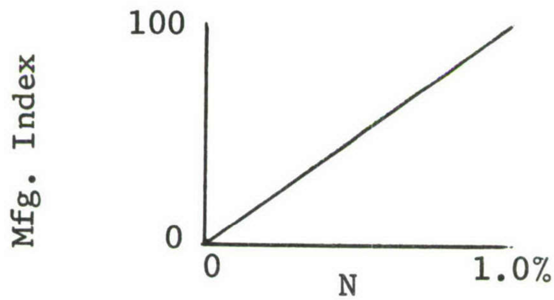
o Inspectability (5%)



	Index
- % structure accessible for inspection	0 - 10
- Capability & accuracy of inspection techniques available for particular structural config.	0 - 80

	Index
- Level of operator competence & time required to inspect	0 - 10
TOTAL INDEX	0 - 100

o <u>Manufacturability</u> (1%)

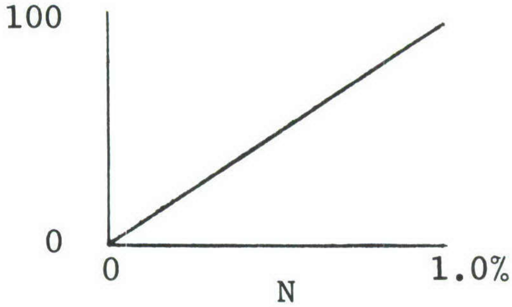


	Index
- Major Processes Required Degree of difficulty and time to accomplish as related to a factor of 1 for conventional machined-bolted/ riveted aluminum structure -	0 - 80

Machining
Welding
Etching
Adhesive Bonding
Brazing
Stretch Forming
Close Tolerance Bolts
Interference Bolts

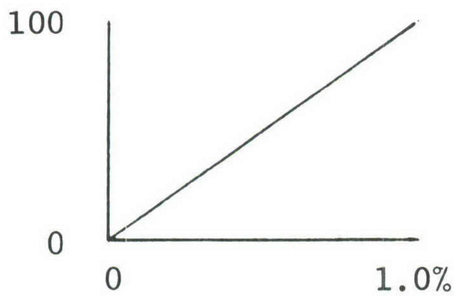
- Relative degree of difficulty to implement and maintain adequate quality control	0 - 20
TOTAL INDEX	0 - 100

o Maintainability (1%)



	Index
- Access to structure for required maintenance actions . Quick versus bolted structural doors . % of structure accessible	0 - 40
- Periodic maintenance requirements such refurbish corrosion protection systems, fuel seal systems, replace access door bolts or quick-release mechanisms	0 - 30
- Damage susceptibility of the structure	0 - 30
Total Index	0 - 100

o Repairability (1%)



	Index
- On-Board" repair capability for most probable type of failures in non-critical structural elements	0 - 20

	Index
- On-Baord" repair capability for cracks in critical structural elements for which damage tolerant design is being provided such as:	0 - 80
. crack through and across a laminate	
. Partial de-hand or braze line failure (Exclude complete element failures which are extensive enough to have caused loss of the airplane if the structure had been designed conventionally).	
Total Index	0 - 100

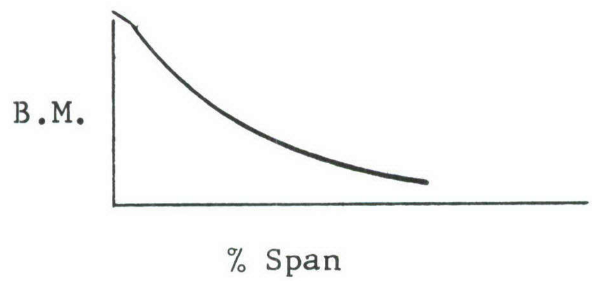
o Predictability (1%)

	Index
- Structural arrangement is such that currently recognized stress analysis techniques, such as finite element analyses will predict internal loads to a level of accuracy consistent with current in-use safety factors for static strength and scatter factors for fatigue strength.	0 - 80
- Behavior of the total structure can be predicted following postulated failure of single structural elements.	0 - 20
Total Index	0 - 100

Several parametric studies are already planned for Phase Ib. These will include the effect of change in time at temperature on the designs and the effect of a change in the maximum temperature. Another trade study will involve the application of fatigue/fracture control criteria as follows:

- o What weight/cost penalties are associated with application of fracture control criteria from wing tip to fuselage?

- o Can a stress related cut-off be established which permits criteria relaxation with no increase in risk? Known risk? Maybe B.M. curve reflects variable criteria curve



- o Outboard wing design criteria

  - . Removable/replaceable/throw-away parts?
  - . Very repairable parts?
  - . Modular removable unit - factory refurbishable?

3.1.5 Computer Programs - During this or subsequent phases of the program, should computer program(s) be used other than those already in existence, they will be made operational in accordance with R&E-15-FDL-2, on the ASD computer at WPAFB, Ohio, and will become the property of the government. The effort to satisfy paragraph 1.b.(2)e. of R&E-15-FDL-2 will not exceed 10 working days on site at WPAFB. Government personnel will be instructed in the use of the computer program(s).

TABLE I RATING SYSTEM** FOR THE AIR SUPERIORITY FIGHTER WING STRUCTURES PROGRAM

STRUCTURAL EFFICIENCY = 0.3	TECHNOLOGY ADVANCEMENT = 0.3	INTEGRITY AND RELIABILITY = 0.3	ABILITIES = 0.1
Cost = 0.5	Concepts = 0.3	Static = 0,1	Inspectability = 0.5
Weight = 0.5	Manufacturing = 0.3	Fatigue* = 0.3	Manufacturability = 0.2
	Materials = 0.3	Safecrack = 0.3	Maintainability = 0.1
	Fracture = 0.1	Fail Safe = 0.3	Repairability = 0.1
			Predictability = 0.1

* Service Life maintained at 4000 flight hours. Any design not maintaining this life will be considered unacceptable.

** Revised rating system Jan 1973.

3.1.6 Testing to Support Preliminary Design - Coupon and panel tests shall be conducted as required to develop unavailable design data necessary for the preliminary design and trade studies. It is essential that all mechanical property testing methods and results be presented in a format useful in the design of future Air Force Systems. Guidelines for the presentation of such data have been approved or are under consideration by MIL-HDBK-5. Consequently, General Dynamics will insure that data will be presented to meet these guidelines. Where approved guidelines are not yet available, such as for fracture toughness and crack propagation, General Dynamics will assure that techniques for presentation are compatible with the currently proposed MIL-HDBK-5 guidelines. Additionally, all of the material property test data generated will be presented as a separate item and in a format which can be submitted directly for the incorporation into MIL-HDBK-5. The applicable portions of the property data generated shall be made available for incorporation into the "Damage Tolerant Design Handbook" being prepared by the Air Force. A plan for testing to support preliminary design shall be prepared for the approval of the Air Force Procuring Contracting Officer.

3.1.6.1 Material Evaluation Testing - Material evaluation will continue in the follow-on program. Emphasis will be placed on the characterization of materials for the specific design requirements of selected configurations, construction, and testing to ascertain compatibility of the joining methods, sealants, adhesives, finish requirements, etc. Close coordination will be maintained among engineering, tooling, manufacturing and inspection personnel in material selection.

Evaluations will be performed to establish the relationship of low-risk to high pay-off status. Results from the materials test and analysis will provide a technological data base that is adequate for acceptance in a systems program with minimum additional development.

3.1.6.1.1 Specimen Test Procedures - Fracture data will be validated on a statistical basis in accordance with the procedures set forth in the 1968 ASTM Standards, Part 31, entitled "Proposed Method of Test for Plane Strain Fracture Toughness of Metallic Materials;" the 1970 ASTM Standards Test Method E399-70T; AFFDL-TR-69-111; or by alternate methods approved by the AFFDL.

3.1.6.1.2 Material Control and Testing - The materials to be used in the follow-on program will be controlled by a system of procedures and/or specifications sufficient to preclude the use of materials with inferior K_{IC} values in fracture-critical areas of the design. Tests will be conducted on all billets, forgings, extrusions, plates, and other forms where thickness permits to evaluate plane strain fracture toughness. Slices will be cut from these items at receiving inspection to provide test specimens. These specimens will be heat treated with the same material from which they were cut.

3.1.6.1.3 Statistical Adequacy of Test Program - A materials test program designed to supplement the preliminary test program and to establish design allowables with a higher level of statistical confidence will be conducted. The test plans will be subject to review and approval of the Air Force Project Engineer.

Consideration will be given to the problems and costs of testing for various parameters with selective emphasis on statistical adequacy. Results from other Convair Aerospace test programs will be included in the statistical data base when these programs meet the testing standards of this program, thus minimizing the program test costs.

3.1.6.2 Design Development and Validation Tests - Plans for development and validation tests include the static, fatigue, and damage tolerance tests described in the following paragraphs.

These tests will provide a sequence of design, development, and validation which is consistent with that proposed in MIL-A-8867. The Phase IA effort consists of the analytical development and validation of the selected wing design configuration(s) based on existing and advanced technologies. The follow-on program provides the opportunity for further optimization of the Phase IA configurations using an appropriate development and validation test program.

3.1.6.2.1 Development Tests of Structural Elements - Structural elements representing the selected Phase IA configurations will consist of such things as laminated panel arrangements including attachments, spars with splices or cutouts, skin panels with access or pylon cutouts, and panels with stringers bonded or mechanically attached.

3.1.6.2.1.1 Static Tests - Tests of elements representing structural configurations established in Phase IA are envisioned as items of primary importance at the start of a follow-on program because of the unconfirmed nature of the preliminary designs made previously.

Tests will be made to confirm the features of the prime candidate and alternates for detail design. These tests will be of the simplest possible nature required to substantiate the design assumptions used in spars, panels, and fastenings. Each type of test will be planned when possible, for performance in a universal testing machine in order to minimize the need for test fixtures.

Test objectives will be to ascertain that the computed design allowables that will reduce weight are met or exceeded and that all the variables that might affect the results of the test or a subsequent component are addressed. Some of these variables are described in the following paragraphs.

1. Shear resistant behavior

2. Tension field behavior

3. Integral construction (extruded, forged, cast or machined)

4. Buildup by fastener attachment

5. Buildup by adhesive bonding, diffusion bonding, brazing or welding

6. Various materials, singly or in combination

7. Location in wing (inboard, outboard, front fuel boundary, aft fuel boundary or intermediate)

8. Special load introductions and splices

9. Cutouts

10. Environment

11. Effects of fuel pressure

12. Variations in dimensions within limits of tolerance

13. Manufacturing variables

14. Quality control variables

Plans for static tests of skin panel elements will include consideration of the following variables:

1. Boundary conditions for stability

2. Integral construction (machined or extruded with integral reinforcement and etched)

3. Buildup by bonding, brazing, diffusion bonding welding or fastener attachment of stiffeners

4. Various materials, singly or in combination

5. Location on wing (upper or lower surface inboard or outboard location)

6. Fuel pressure

7. Cutouts and reinforcements

8. Environment

9. Variations in dimensions within limits of tolerance

10. Special load introductions and splices

11. Extremes of combined loading applications.

At least the following variables will be considered in plans for static tests of joint elements:

1. Adherend materials

2. Adhesive materials

3. Mechanical fasteners

4. Type of joint (permanent or removable for access)

5. Efficiency of joint (not fully effective at low load if removable and in vicinity of permanent joint)

6. Effects of permanent deformations

7. Environment

8. Shear applications

9. Tension applications

10. Lugs

11. Effect of load reversals

12. Mixed fastener patterns

13. Prying or peeling resulting from eccentricity

14. Manufacturing variables (quality of hole finishes, concentricity of holes, uniformity of preload application and tolerances

15. Quality control variables (typical defects permissible by Quality Control).

The results of these first tests are expected to lead to reiteration of some earlier trade-off studies and further tests for the same purpose before proceeding to more comprehensive testing.

3.1.6.2.1.2 Fatigue Tests. Development fatigue tests of structural elements contining critical and/or advanced design details will be planned to determine experimental stress concentration factors (K_f). Sufficient test specimens of each detail will be run so that average fatigue life can be determined and associated K_f factors established. Testing will be conducted using a condensed version of the flight-by-flight fatigue spectrum.

Comparative fatigue tests of structural fastening systems, bonding systems, brazing and welding systems, etc., will be used to establish optimum methods of production. Sufficient specimens will be planned to provide a statistical evaluation of variability.

3.1.6.2.1.3 Damage Tolerance Tests. Development tests will be planned to establish reliably the residual static and fatigue strength of the selected Phase IA configurations. These configurations are envisioned to involve structural elements with and without crack stoppers and redundant structure utilizing multiload path features such as

laminates. Residual strength will be related to limit load (or maximum spectrum load) and to the types and sizes of damage. Damage will include consideration of complete failure of single stringers or laminates, partial failures (flaws or cracks), and the partial consumption of conventional fatigue life prior to flaw introduction. The development of cracking modes, the evaluation of inspection and detection techniques, and crack propagation measurements will all be considered in sufficient detail to allow incorporation of design refinements into the preproduction component tests described in paragraph 3.6.2.

3.1.7 Reporting - At the conclusion of the Preliminary Design Phase, a Preliminary Design Technical Report is required. In the report the two (2) most promising candidates, picked from the three (3) previously evaluated, will be recommended for the Detail Design Phase. The rationale and trade study results on which these decisions are based will be summarized. The final selection of the two (2) most promising candidates will be made by the Air Force Contracting Officer within thirty (30) days of the receipt of the draft of the Preliminary Design Report. Notice of the selection will be made in writing.

3.2 Phase II - Detail Design - A detail design and analysis will be performed on each of the two wing box configurations selected from the Phase Ib effort. The detail design will proceed in accordance with the Detail Design Section of the Master Planning Document.

3.2.1 Strength Requirements - All strength, fatigue and fracture requirements of the baseline aircraft box structure, as provided in MIL-STD-1530, shall be satisfied. Critical areas of the structure shall be indicated, and a positive margin of safety will be established for all members. Areas critical in fatigue or fracture will also be indicated and the respective fatigue life inspection interval predicted.

3.2.2 Functional Similarity - All functional aspects of the baseline wing box structure, including appropriate joint and splice provisions, will be included in the detail designs. Provisions will be made, as applicable, for such items as outboard-wing attachments, landing-gear attachments, landing gear doors, bomb bay and bomb bay doors, fuel bay and fuel bay access openings, internal structure inspection, and other access requirements as dictated by the baseline vehicle.

3.2.3 Dimensional Constraints - The outside dimensional envelope of the baseline vehicle will be maintained.

3.2.4 Test Provisions - The selected configurations will incorporate static and fatigue test load attachment fittings as required. The design of these fittins will be subject to the approval of the Air Force Contracting Officer. The structure of the baseline vehicle, adjacent to the wing box will be simulated to the extent necessary for proper introduction and stress distribution in the area of interest in test article. A preliminary test plan will be prepared for the full scale testing in Phase IV.

3.2.5 Drawings - A set of engineering drawings suitable for production release will be prepared on 0.003 mylar for each of the selected configurations using AF drawing formats. These drawings are subject to the approval of the Air Force Contracting Officer. Fracture critical parts (or zones) will be identified on the drawings in accordance with fracture control plan requirements of MIL-STD-1530.

3.2.5.1 Fracture Critical Parts - A part is defined as fracture critical if catastrophic failure of the part would result in loss of the aircraft.

Critical parts will be selected by a review of primary structure which is principally loaded in tension. Parts experiencing exposure to a corrosive environment will also be reviewed for possible inclusion in the fracture critical parts list.

The review will be a joint effort by the Structural Design and Analysis Groups and the Fatigue and Fracture Analysis Group. The review will result in a fracture critical parts lists which will be updated on a systematic basis as the design evolves. Trade study results, such as initial damage sizes, will be reflected as they become available and revisions made as necessary to the parts list.

The critical parts list will provide the following information for each part as a minimum:

1. Part description and location in the structure

2. Drawing number

3. Type of material and basic form

4. Type of fabrication applied to the part, if any.

The critical parts list will be maintained and updated as required by the Structural Design and Analysis Groups. The list will be distributed to supporting groups and reissued as revisions are made.

3.2.5.2 Design Drawings for Fatigue and Fracture Control Parts - The engineering drawing is the single means of transmitting the requirements of the fracture control plan. Fracture critical parts will be identified by the following drawing note:

This part is categorized as a fracture critical part and is subject to all requirements of the fracture control plan.

Material procurement and material processing specifications along with NDI and corrosion protection requirements will also be specified on the drawings for all critical parts. No deviation from these drawing requirements will be permitted without approval from the Air Force Contracting Officer.

Drawings in which only portions of the part is categorized as "fracture critical" will be zoned to identify these areas. These areas will refer to a note on the face of the drawing which will read:

This zone of the part is categorized as a fracture critical zone and is subject to all requirements of the fracture control plan.

Fracture critical parts processed in accordance with toughness controlled specifications will include a test tab for certification of fracture toughness subsequent to processing. The drawing will identify and locate the test tab.

Typical drawing notes for a fracture critical part are as follows:

1. This part (zone) is categorized as a fracture critical part (zone) and is subject to all requirements of the fracture control plan.

2. Serialized traceability is required.

3. The material must meet the special requirements of

4. Braze (bond, weld, etc.) per

5. Special corrosion protection required per

6. Perform NDI in accordance with

7. Fasteners shall be installed and inspected in accordance with

During preliminary and production design, Quality Assurance will review drawings to ensure that all inspection and maintenance requirements are documented in a Preliminary Q.A. Plan.

3.2.6 Cost and Weight Estimates - Detailed projected production cost and weight estimates for each design considered in this Phase will be established. In addition, cost/weight control will be exercised throughout the program. As appropriate for alternative design selections, life cycle cost considerations will be analyzed. Production buys of 1, 2, 50, 200 and 800 ship sets will be utilized in the cost projection. Documentation will include the basis and rationale for computations including production rates, time period, etc. These estimates will be based on a Preliminary Manufacturing Plan prepared by Manufacturing Engineering.

3.2.7 Trade Studies - Expansion and refinement of the trade studies performed under Phase I will be accomplished. These studies will be conducted on each configuration and will be based on findings of all previous efforts.

3.2.8 Testing to Support Detail Design - The results from the development tests of structural elements will be incorporated into preproduction test components. These test components will be a series of box beams representing final or near final structural designs.

3.2.8.1 Static Tests - A minimum number of box beam tests will be required after careful screening of the earlier element test results. This type of test, however, will be imperative for confirmation of the stability and overall behavior of the configurations which show the highest probability of effecting a weight reduction with respect to the baseline and also appear satisfactory in terms of cost and durability.

Plans for these tests will be firmed after considering the optional combinations of the more attractive results of the element tests. The same variables that were considered in the element tests will be considered at this point when applicable.

The objective of these box beam tests will be to establish the highest possible design allowables for combined loadings (bending about two axes, torsion, shear along two axes, internal and external pressure) and thermal effects. Major splices will be simulated.

It is possible that repetitions of these box beam tests will be required to ascertain the improvement of overall behavior obtained by changes to some of the design features.

When possible these test articles will be designed to be representative of the wing inspectability so that inspection can be made during the progress of tests for possible prevention of overall failure. In this way, plans may be made for repetitions of tests after modifications.

3.2.8.2 Fatigue Tests - A number of box beam tests will be planned to validate fatigue resistance. Testing will be conducted using a flight-by-flight, random-cycle ordered fatigue design loads spectrum that will include compression loads. The effects of environmental conditions (pressure, temperature, and chemical) will be considered.

A sufficient number of wing root-splice specimens of final or near final design will be fatigue-tested to provide K_f data and a basis for reliability analysis.

3.2.8.3 Damage Tolerance Tests - Box beam tests will be planned to validate the damage tolerance of the selected wing design. Fail-safe evaluation will involve severing stiffeners, laminates, or spars and determining the residual strength. Slow crack growth capability will be validated by the introduction of flaws as required by the criteria into skins, stringers, fastener holes, etc., and applying a random-cycle ordered, flight-by-flight fatigue spectrum identical to that used for fatigue safe-life testing.

Wing root splice specimens will be flawed in the critical fastener holes and both static and fatigue tested. Crack growth rates will be monitored throughout the damage tolerance tests.

3.2.9 Test Fixture and Jig Design - The contractor will design all test fixtures and jigs necessary to conduct the full scale static and fatigue tests required in Phase IV. Assume utilization of two existing wing holding fixtures at CA/SD, and redesign and modification of existing whiffle trees (as feasible). Design loading fixtures for applying loads at the slat and flap attach points (positive and negative loads) and the pylon hardpoints (negative loads). Perform fatigue analysis of holding fixture previously used for fatigue testing to assure remaining life sufficient to apply six service lives of fatigue testing to the fatigue test wing.

Design test setup for both static and fatigue test articles including the requirements for load programmers, load control servos and data recording equipment.

3.2.10 Reporting - At the conclusion of the Detail Design Phase, a Detail Design Technical Report is required. In the report, the single most promising configuration, picked from the two (2) previously evaluated, will be recommended for fabrication by the contractor. This recommendation will also be made in a design review to be held at the contractor's facility. The rationale, trade study results, and design data on which these recommendations are based, will be reported. The final selection will be made by the Air Force Contracting Officer within thirty (30) days of the receipt of the draft of the Detail Design Report. Notice of the selection will be provided to the contractor in writing. Approval to initiate long lead time procurement items in advance of the final design selection must be requested from the Air Force Contracting Officer in writing.

3.3 Phase III - Fabrication - Two (2) identical full-scale wing box structure articles will be built to the engineering drawings for the final configuration developed in Phase II. An alternate plan to fabricate only one full scale wing box will also be considered. The fabrication shall proceed in accordance with the Fabrication Section of the Master Planning Document; findings of previous effort will be used to revise, as appropriate, the Fabrication Section before the start of this phase. Manufacturing methods which promise the greatest return in terms of reliability, integrity, and efficiency will be employed. This Phase will be conducted in a manner such that design related problems, that become known during fabrication, can be identified and resolved. A primary objective of this Phase is to minimize

those problems that would later impace a full production schedule. Quality assurance problems will be identified and corrected. This phase will be documented in detail.

3.3.1 Test Fixtures (reference paragraph 3.2.9) - The contractor will fabricate or provide all test fixtures and jigs necessary to conduct the full scale static and fatigue tests required in Phase IV.

3.3.2 Instrumentation - During assembly of the test articles, the contractor will install that instrumentation and test fixtures which properly should be installed during assembly.

3.3.3 Documentation - The contractor will provide sufficient documentation to establish that the selected design represents the effective integration of all relevant technology areas. Specifically, Category (C) NDI methods will be demonstrated acceptable to the cognizant Air Force personnel.

3.3.4 Cost and Weight Determination - Manufacturing costs and structural weight estimation techniques will be verified by comparison of predicted and actual values; disparities will be noted; estimation techniques will be re-examined, re-evaluated, corrected and documented.

3.3.5 Fabrication - Fabricate two identical left hand, full-scale F-111 type wing box articles. Wing Pivot Fittings (12W473) have been welded and heat treated and are now ready for finishing and installation into the completed wing box. These pivot fittings are currently on hold in a storage area.

3.3.6 Manufacturing Support - Structural Design and Stress Analysis will provide manufacturing support as required during fabrication of the two test articles.

3.3.7 Materials Traceability - In accordance with the fracture control plan requirements of MIL-STD-1530, the following will be accomplished for each fracture critical part, complete data documenting the raw material heat number, manufacturing planning, inspection records, discrepancy reports will be recorded, collected and maintained. These records will provide complete traceability of produce quality from raw material through the completed assembly. Traceability will be implemented in accordance with Standard Practice 9-23.1.

In order to race raw material through all processing, the vendor heat or lot number is related to the first shop order serial number and part number at first cut level. The shop order serial number identifies the part through to the end item.

3.3.8 Final Test Plan/Instrumentation - Prepare the final test plan for the Phase IV testing and submit for customer approval.

Define strain gage locations for each of the test articles and install the internal strain gages during fabrication. Install external strain gages after completion of fabrication. Assume 200 strain gages required for the static test article, and 150 strain gages for the fatigue test article.

3.3.9 Spares - The contractor will identify and reserve not less than 10 percent of the Phase III funds to be used for the fabrication of spare parts. These parts will be fabricated, as required, to replace parts damaged during the test program. In the event of premature failure of a part(s), these funds will be used to redesign and fabricate a suitable replacement part(s). Modification, replacement, and repair of parts of the test articles will be made only with approval of the Air Force Contracting Officer.

3.3.10 Production Quality Assurance - A final Quality Assurance Plan will be prepared for the configuration selected for Phase III fabrication. This plan will stipulate the inspection, tests, process controls, and data recording tasks for purchased materials, fabrication processes, and final acceptance. It will describe what controls (e.g., inspections, tests, etc.) will be applied, when they will be applied, the procedure to be used, and the criteria for acceptance.

Quality Assurance personnel will be trained and qualified in the use of any new procedures or methods such as NDI.

During fabrication, the application of the planned quality controls will be closely monitored. Positive action will be taken to correct deficiencies revealed in the hardware or documentation.

Quality Assurance records will document the complete history from raw material through final inspection. A summary of these records and appropriate analysis will be included in the fabrication report.

3.3.11 Fabrication Report - At the conclusion of all fabrication, modification, and repair of the test articles, a Fabrication Report prepared by Manufacturing Engineering is required. Detailed coverage of the fabrication phase is required with particular emphasis on information useful for planning production of a future aircraft.

3.4 Phase IV - Test and Evaluation - The structural strength of the final selected design configuration will be demonstrated by full-scale static, fatigue, and damage tolerance tests on the wing box. These tests will be designed to provide proof of compliance with the design criteria. Preliminary test planning for the follow-on program will be accomplished in the second contract phase (Phase II). Preliminary test planning will include a general description of the test article(s), instrumentation requirements, test loading and inspection requirements, and test procedures. Preparation of the final test plan, manufacture of the test article(s) and test fixture(s), and test performance will be accomplished in Phase III of the follow-on program.

The scope and general definition of the follow-on program for the full-scale articles, as currently envisioned, are described in the following paragraphs. A baseline test program that includes fatigue testing, ultimate static testing, and damage tolerance testing on two separate test articles will be considered, initially. During the initial contract phase, studies will be made on cost-saving alternate test programs that would yield acceptable results. Alternate test programs to be evaluated include the following:

- Feasibility of conducting damage tolerance tests on the static test article, fatigue test article, subcomponent articles or combinations thereof
- Cost savings realized by using a single test fixture
- Possibility of deleting ultimate static tests
- Possibility of cold proof tests for noninspectable areas.

3.4.1 Full Scale Static, Fatigue, and Damage Tolerance Tests - The test article will consist of a structurally complete wing box of the selected design including a wing pivot fitting the same as (or similar to) the F-111F wing pivot fitting to facilitate loading (including adaptation to current fixtures). Flap and slat loads will be introduced (as required) through load fittings at the correct load points.

The baseline test program outlined in the following paragraphs is based upon the assumption that two separate test articles will be used (static test article and fatigue test article). Alternate programs will be fully developed in the follow-on program.

3.4.1.1 Static Tests - The static test article will be tested at room temperature to 150 percent of design limit loads. Approximately three conditions will be required to demonstrate compliance with the static strength design criteria. Critical loading conditions will include two positive load factor conditions and one negative load factor condition. The positive condition loads will be applied utilizing upper surface load bolts, high-lift attachment point fittings, and wing hardpoints. Negative condition loads will be applied using high-lift attachment point fittings and wing hardpoints.

The test article will be thoroughly instrumented to determine stresses and deflections. Approximately 200 channels of instrumentation will be required. A Structural Acoustic Monitoring (SAM) system, which has proved very useful as an inspection aid in the F-111 cold-proof test program, will also be utilized in the static test program.

Inspection requirements will include pretest inspections as well as posttest inspections to establish initial and final status of the structure. Inspection procedures will be selected from the most up-to-date state-of-the-art techniques applicable to the test structure.

Three fail safe tests will be performed on the static test article, each with a precracked or severed structural member to represent critical damage conditions for the purpose of demonstrating the fail safe characteristics of the wing. These tests will be performed in accordance with the requirements of MIL-STD-1530, MIL-A-8866A (including proposed revision providing requirements for damage tolerance

testing) and MIL-A-8867A (USAF). The test conditions will be the same as those used for the ultimate static tests. Test load levels for each test will be approximately limit load.

Selection of the critical areas for these tests will be based on analyses and tests performed in the Phase Ib and II development programs.

Repair of each intentionally damaged area is required after completion of the fail-safe test and prior to the next test. After the insertion of the intentional critical damage, the areas so treated will be reinspected by NDI in order to provide records of the damage as it appears to NDI before the fail safe tests. After the test, the test article will again be subjected to NDI in order to determine the extent of change or growth in the damage caused by the test.

3.4.1.2 Fatigue Tests - The fatigue test program will be designed to demonstrate a safe-life goal of one service life with a scatter factor of four. The proof of compliance with the design fatigue load criteria will consist of fatigue testing the full-scale fatigue test article to the requirements of MIL-A-8866 and MIL-A-8867. The load factor exceedance data to be used for the service loads development, fatigue analysis, and test spectra development will be based on the requirements of MIL-A-8866. The test spectra will be derived for the F-111 service usage using the mission analysis approach.

Testing will be conducted at room temperature on a flight-by-flight basis. Two test load distributions will be used, one positive and one negative. Strain gages will be installed in high stress areas significant load paths, and areas of stress concentration as determined from preliminary stress and fatigue analysis. These gages will be monitored on an intermittent or continuous basis, as required, to provide information for failure detection, fatigue analysis verification, and load distribution verification.

Inspection requirements will include pretest inspections as well as periodic nondestructive inspections performed at key points in the test program to determine if testing has developed any flaws. Inspection procedures will utilize the most up-to-date state-of-the-art techniques applicable to the test structure.

3.4.1.3 Crack Growth Rate Testing - Damage tolerance testing will be conducted on the fatigue test article after the four lives of testing is complete.

The purpose of the damage tolerance testing on the full-scale article will be to show that the selected design meets the damage tolerance criteria of MIL-A-8866 and the proposed MIL-STD-1530 (USAF) "Aircraft Structural Integrity Program (ASIP): Airplane Requirements," dated May 1972. The proposed testing will include fatigue loading to verify the analysis crack propagation rates. These tests will consist of preflawing the structure in multiple areas (assume four) and conducting fatigue testing until failure occurs (or maximum of two service lives). Inspection requirements will include nondestructive inspection prior to and after flawing the structure and periodic inspections performed throughout the test program to monitor flaw growth. Preflawing techniques will be included in the final test plan development during the follow-on program.

3.4.2 Test Fixtures - Considerable cost savings can be realized by using existing wing holding fixtures for the selected test program. The structurally complete wing box will be installed in the wing holding fixture used for the qualification tests of the F-111 wing. One such fixture currently exists at AFFDL's Structures Test Facility (under F-111 Contract ECP 10212); and two such fixtures, previously used in the static and fatigue qualification tests of the F-111 wing, exist at the Convair Aerospace Division at San Diego, California. The test setup will be similar to that shown in Figure 3-5. Consideration will be given in the preliminary test planning to conducting all or part of the testing at AFFDL's Structures Test Facility. This could very well be the most economical approach if the Air Force is willing to preserve the previously mentioned test fixture for this purpose.

3.4.3 Test Report - Strain and deflection data from the full-scale structural verification tests will be reduced to engineering units by electronic computers. Computers will also be used to plot the results of selected critical strain gages. The use of computers will provide rapid dissemination of the structural test datato all interested agencies.

3.4.3.1 Analysis and Evaluation of Test Results - The data from the full-scale structural tests will be analyzed and compared with predicted results based on analyses performed earlier in the program. The static test results will be used to determine stresses and deflections and to verify load paths and analytical methods. Data from the final intentional failure test will be used to verify predictions and to evaluate growth potential. Fatigue test data will be used to estimate the service life and to determine inspection interval requirements. Data from damage tolerance tests will be used to verify analytical methods and, in conjunction with fatigue tests and fracture mechanics analyses, to determine inspection interval requirements.

Results of the structural verification testing will be analyzed by quality assurance engineers for evidence of inadequacies in the quality assurance plan. Inspections, tests, and process controls related to failures or marginal results will be reviewed, and changes incorporated when required.

3.4.3.2 Contractor Recommendations - Data from the full-scale structural tests will be used by the contractor to provide recommendations for possible changes in design, tests, or fabrication. These recommendations could include design changes to reduce weight or stress concentrations, improved analytical methods, manufacturing methods, and inspection requirements and procedures.

3.5 Information Transfer - Convair Aerospace proposes a three-part program to advance the state-of-the-art for information transfer and data dissemination. The three parts are Part I--Continuous Survey, Part II--Equipment-Media Demonstration, and Part III--Information Transfer Model.

3.5.1 Part I--Continuous Survey - A continuous survey will be conducted for the duration of the program. Military, Government, industrial, and educational institutions will be examined to determine what audio and visual equipment is

readily available, the facilities and operational capability, and the evaluation of information transfer effectiveness versus information media used. Secondly, an extensive survey will be made of all published materials and of research projects either in process or completed on effective information transfer. As the survey progresses, various equipment-media will be used to request, record, transmit, and report information.

3.5.2 Part II--Equipment-Media Demonstration - An effort will be directed toward demonstrating the application of equipment to the transfer of specific types of information such as verbal, graphic, processes, operational, organizational and reference data. Equipment-media to be demonstrated will include, but not be limited to, printed reports, video tapes, audio tapes, slide film, microfilm and motion picture film.

Secondly, recently developed equipment-media will be investigated in Part II to determine application advantages and limitations and the types of information most suited to transfer.

Thirdly, consideration will be given to matching the equipment-media with various sequences and mixes of information types. For example, the type of distribution planned for information may dictate the use of a particular information equipment-media if animation of graphics or flow diagrams are involved. Examples of investigations to be conducted in Part II are:

Information Type	Equipment-Media
Verbal	Video Tape Audio Tape Audio-Slide Film Motion Picture Film Printed Page Microfilm Transparency Telephone
Graphics	Video Tape Slide Film Motion Picture Film Printed Page Microfilm Transparency

Information Type	Equipment-Media

Animated Graphics	Video Tape
	Motion Picture Film
	Technimation--
	  Video Tape
	  Motion Picture Film
	  Slide Film
	  Transparency
	
	Audi-Pointer
	  Video Tape
	  Motion Picture Film
	  Printed Page

3.5.3 Part III--Information Transfer Model - A model for information transfer and data dissemination will be developed. The model will match information-type, recommended equipment and transfer media (advantages and disadvantages), and information use. Particular attention will be directed to material preparation cost and difficulty, equipment costs and acquisition availability, scope of information dissemination and use, cost of supplies and transfer media reproduction, and transmittal methods and time spans. Because of the interdependency of program parts, all three parts will be conducted during the full span of the program.

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13. ABSTRACT

This report describes the preliminary design and analysis for an Advanced Air Superiority Fighter Stores Loaded, Wet Wing Structure. The wing box of the F-111F airplane designed by the Convair Aerospace Division of General Dynamics was used as the baseline vehicle.

A unique design methodology was followed to arrive at three configurations which offer an optimum balance between structural efficiency and technological advancement. This methodology consists of compiling element concepts; integrating them into cross-section drawings; optimizing them in analytical assemblies; and finally preparing full wing box designs. Each step was followed with a detailed evaluation and ranking step which utilized a formal merit rating system. This system permitted the evaluation of numerous concepts and insured that each technical discipline participated in the design selection.

A subsequent program is proposed to evaluate the capability of the selected design to meet the overall program goals of advancing technology without significantly affecting costs. The subsequent program involves additional preliminary design, a development test program, detail design, manufacture, and tests; including static, fatigue, and damage tolerance testing. Information generated during this effort will be disseminated to the Air Force and industry in general through an intensive information transfer effort.

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14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Structural Design						
Stress Analysis						
Fatigue						
Fracture Analysis						
Materials						
Mass Properties						
Value Engineering						
Manufacturing Engineering						
Nondestructive Inspection						
Quality Assurance						

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